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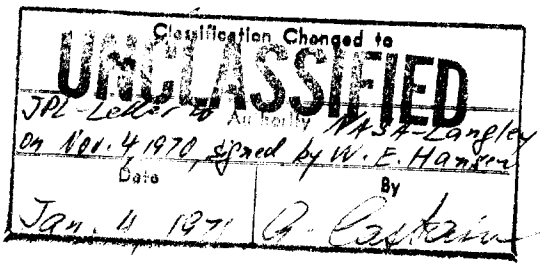
Space Programs Summary 37-43, Vol. I

The Lunar Program

For the Period November 1 to December 31, 1966

(Title Unclassified)



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LAB-DC1C45

January 31, 1967

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SPACE PROGRAMS SUMMARY 37-43, VOL. I

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Prepared Under Contract No. NAS 7-100
National Aeronautics and Space Administration

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Preface

The Space Programs Summary is a six-volume bimonthly publication designed to report on JPL space exploration programs and related supporting research and advanced development projects. The titles of all volumes of the Space Programs Summary are:

Vol. I. *The Lunar Program* (Confidential)
Vol. II. *The Planetary–Interplanetary Program* (Confidential)
Vol. III. *The Deep Space Network* (Unclassified)
Vol. IV. *Supporting Research and Advanced Development* (Unclassified)
Vol. V. *Supporting Research and Advanced Development* (Confidential)
Vol. VI. *Space Exploration Programs and Space Sciences* (Unclassified)

The Space Programs Summary, Vol. VI, consists of an unclassified digest of appropriate material from Vols. I, II, and III; an original presentation of the JPL quality assurance and reliability efforts, and the environmental- and dynamic-testing facility-development activities; and a reprint of the space science instrumentation studies of Vols. I and II.

Approved by:

W. H. Pickering, *Director*
Jet Propulsion Laboratory

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I. *Surveyor* Project

LUNAR PROGRAM

A. Introduction

The *Surveyor A-21* model flight spacecraft (Table 1) are designed to span the gap between the *Ranger* Project and the *Apollo* Project. They will demonstrate the technique of making soft landings on the Moon, extend our knowledge of lunar conditions, and determine the suitability of sites for proposed *Apollo* spacecraft landings.

Table 1. Model and vehicle designations

Designation	Description
SC-1–7	Flight spacecraft with engineering payload
A-21	Model designation for spacecraft with engineering payload (SC-1–7)
A-24	Folmer model designation for spacecraft with proposed scientific payload (SC-8–10)
T-21	Model A-21 prototype system test spacecraft
T-2	Simplified spacecraft for evaluation of terminal descent propulsion and flight control
S-2	Test spaceframe for vibration, shock, and static structural tests
S-6, -7	Test spaceframe for vernier propulsion system tests
S-9	Structural test model
S-10	Temperature control model

Surveyor I, the first flight spacecraft, was launched from Cape Kennedy, Florida, on May 30, 1966, and soft-landed on the Moon on June 2, 1966. By June 14, when lunar sunset occurred, approximately 100,000 commands had been received by the spacecraft; 10,338 pictures of the spacecraft and its immediate vicinity had been transmitted. During the second lunar day 899 additional pictures were received.

Mission B, which initially gave every appearance of the success of mission A, was terminated in the early morning of September 22, 1966. The spacecraft and all other increments of the mission performed nominally up to execution of the midcourse thrust maneuver. At that point, vernier engine number 3 failed to ignite; the resulting imbalance of thrust from vernier engines 1 and 2 imposed a tumbling motion on the spacecraft, from which it failed to recover. Subsequent attempts to fire the vernier engines eventually increased the tumbling rate to 146 RPM. The main retro-rocket was ignited and burned for approximately 30 seconds, at which time contact was lost with the spacecraft. At 2:53 a.m. PDT, following continuous receiver search for the spacecraft with no results, mission B was terminated.

Because of the high degree of success of the mission up to the point of the midcourse thrust maneuver, data of considerable value were derived. The full significance of the transmitted flight data, as well as failure data, is

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still undergoing analysis and interpretation and has already contributed valuable guidance for *Surveyor* mission C which is scheduled for the first quarter of 1967.

Hughes Aircraft Company (HAC), Space Systems Division, is under contract to fabricate the *Surveyor A-21* spacecraft. The launch vehicle, a combination *Atlas/Centaur,* is provided by General Dynamics/Convair. Control, command, and tracking functions for the *Surveyor* missions are performed by the JPL Deep Space Network and Mission Operations System.

B. Systems Engineering

1. Failure Review Board Analysis

Several meetings have been held to discuss SC-2 anomalies reported by the SC-2 failure review board in order to ensure that all steps are taken to prevent a recurrence of any anomalies that occurred during the SC-2 mission. A formal failure review board report will present all available details regarding anomalies that may represent contributions to the prime SC-2 failure. SC-2 anomalies of interest are:

(1) Anomaly 11, concerning the inadvertent switch to inertial mode at the legs-down command. The flight control system should have remained in the rate mode until 51.2 sec after separation. However, the flight control switched back to rate mode at spacecraft separation as it is programmed to do, and sun acquisition proceeded normally. It is believed that this anomaly did not contribute to the SC-2 inflight failure at mid-course, but, to assure that this type of failure would not cause any other problems, it is recommended that the gas jet 22-v power be interlocked with post-separation, sensing, and arming 22-v power so that if, for some reason, the gas jets are enabled before separation, they cannot operate, i.e., cannot use nitrogen. ECR 354849 will implement this recommendation.

(2) Anomaly 13, concerning Canopus intensity being outside the upper threshold limit. Test plans are being redesigned for SC-3 to ensure that the Canopus sensor sun filter is selected to ensure automatic star acquisition. Anomaly 14 concerned a higher than predicted background illumination that appeared on the Canopus sensor output. If the Canopus sensor exceeds specification requirements, lower-level background stars will be seen as noise. Attempts are being made to determine the sensitivity of the sensor and to correlate it with the actual star levels noted during flight.

(3) Anomaly 4, in which unregulated current (EP-4) was noisy during and after the midcourse and 20-sec burn. It was generally agreed that there were cyclic loads on EP-4, making the noise normal data. In anomaly 4, the solenoid operating valve (SOV) unregulated current value (EP-4) Δ was lower than calculated. It was also low in performance verification test 5.

2. TV Auxiliary Mirrors

As planned, SC-3 and SC-4 flat mirrors have been received from the vendor, Tinsley Laboratories. The mirrors are now being tested for reflectivity, and will be thermally coated prior to installation on the spacecraft. A mechanical-fit check will be performed on the SC-3 spacecraft prior to shipment.

A study to evaluate the advantages and disadvantages of convex, spherical mirrors in comparison with flat mirrors has shown that no immediate advantages can be gained through the use of curved mirrors.

3. Lunar Hopper

By JPL request, a hopper capability will be added to the *Surveyor* spacecraft to provide for lunar translation after initial touchdown. Short-hop capabilities and constraints have been defined for SC-3. SC-4 capabilities are expected to be similar to those of SC-3.

JPL change order 137 provides for short translation to be accomplished by refiring the vernier engines, causing a sufficient translation to provide a different view of the lunar surface. Since it may be desirable to leave the elevation axis locked, and since it may not be necessary to use it during initial landing and interrogation, a separate unlock for the roll and elevation axes was incorporated. This was the only major change required for the short translation capability.

Long hopping capability for SC-5 and following spacecraft is still being studied for effects and cost. It requires the addition of contact switches to shut off the vernier engines when the spacecraft resettles on the surface and the addition of an antenna/solar panel positioner (A/SPP) relock to prevent possible damage during translation. Long translation will be accomplished by programming a certain thrust for a certain length of time. Exercising control through the flight control system raises the possibility that the spacecraft might continue hopping, because of the time delay involved in transmitting a signal to shut off the vernier engines, and since the crushable blocks may be damaged during initial impact, they cannot

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be depended upon to absorb any further shock. Engines will be left on at 0.9 *g* until the spacecraft settles on the surface again at which time contact switches will immediately shut off the engines.

4. Radar Altimeter and Doppler Velocity Sensor (RADVS) Engineering

a. RADVS signal data converter parallel design (ECP-10). A parallel effort to redesign the RADVS signal data converter, presently under way, will provide an additional capability to differentiate between the desired doppler velocity signals and interference arising from vibration effects. Mechanization is accomplished by using double sideband discrimination and sampling the signal-to-noise ratio in the acquisition control circuit.

b. RADVS crosscouple sidelobe logic (ECP-23). Analysis of system characteristics and performance shows that some possibility of crosscoupling exists between the various main beams and one or more of the sidelobes that are not now covered by circuit rejection logic. A development program is now under way to provide rejection logic for beams 1 and 2 and beams 1 and 3.

Although ECP-10 and ECP-23 were originally initiated as two parts of the same technical effort, a higher priority has since been placed on ECP-23.

c. Circuit design. The 20 different types of submodules for ECP-10 trackers have been tested for design confidence prior to the start of motherboard testing. Only minor problems were encountered, providing a high confidence factor for motherboard testing.

d. Circuit analysis. Equations were written to determine the effects of preamplifier and noise slope on the tracker output frequency at initial acquisition. Actually, only the differential between the noise slopes of the main beam and crosscoupled beams is of interest to the crosscoupled sidelobe logic, because it is this differential that causes a differential frequency in the two beam tracker frequencies. The equations provide for this differential effect. These equations are being programmed for the IBM 360 in Fortran IV.

e. Determination of crosscoupled sidelobe beam angle. Angular shift in the return signal of a transmission mainlobe and a sidelobe skirt of a receiver was calculated from theoretical considerations and graphically determined from measured antenna patterns. Results from both methods were in reasonable agreement. The mathematical value of directional shift was 0.25 deg, and the graphical method yielded a 0.17-deg angular shift from the transmission mainlobe.

f. Product design. All manufacturing drawings for ECP-23 crosscoupled sidelobe logic have been released with a final parts count of 732.

Upgrade of the ECP-10 Y-2 signal data converter to flight configuration is dependent on the resolution of the 2N4042 transistor problem. Reliability of this part is in question because of a glass-lead seal problem. Product design is investigating use of the 2N4878 transistor, which is electrically the same as 2N4042 but is packaged in a TO-18 case configuration.

5. HDM-1E, HDM-1E/M Microwave Diodes

Investigation of the RADVS microwave diode problem has been completed by the *ad hoc* committee consisting of representation from Ryan, JPL, and HAC/Newport. It was concluded by the committee that sufficient insight into current diode problems has been gained during the phase I portion to warrant dissolving the committee and terminating its activity. A final report will be prepared to summarize the results of the analysis conducted.

6. Spare VA515C Klystrons

Of the last eight klystron tubes delivered by Varian, three were usable. Although all tubes met the requirements of specification control drawing 400553, a completely unpredictable vibration phenomenon was present in some units, making them unsuitable for use in the RADVS klystron power supply modulator (KPSM). This characteristic was recognized last June, and work with Varian to investigate the problem was begun. The probable cause has been isolated to the area of the cathode structure; in view of this, the following action has been taken:

(1) One noncompatible tube was rebuilt with a new, tightly fitting cathode heater assembly. This resulted in a compatible tube with regard to vibration response characteristics.

(2) A build of three tubes with random tightness of the cathode heater fit was completed. One of the three was compatible with the KPSM.

(3) A build of two tubes with an intentionally tight-fitting cathode heater assembly was completed, with the result that one of the two tubes was compatible.

Specific structural differences that make some tubes noncompatible with the system are not precisely known at this time.

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7. KPSM Ripple as a Function of Input Voltage

During acceptance testing of SN-11 KPSM, ripple voltage in the velocity sensor channel rose to amplitudes well above the tracker "lock-up" level when excited discretely at 6.5 kHz by a frequency of conversion reached when the preregulator switching the reactor is vibrating sympathetically with the 6.5 kHz tube resonance. The excitation materializes when the input voltage v_i is varied from $v_i = 16.5$ v to $v_i = 22.5$ v.

To reduce this signal to a safe level it was found necessary to:

(1) Reduce the effective harmonic content of the switching current leading edge, as supplied through L-1.

(2) Change the characteristics of the velocity sensor corner section.

To slow the rise rate of the switching currents, a small positive feedback was introduced into the preregulator shunt regulator by moving the feedback lead (located between the preregulator submodules) until a small oscillation of approximately 1 mHz occurs. This action allows high frequency current to flow in L-1 during the switch-on, switch-off period, thus "softening" the surge into L-1. This, in turn, reduces the mechanical and acoustical excitation to V-1. Attenuation attributed to this action was approximately 40 db and was marginally close to resolving the problem.

A cracked corner section in the KPSM also contributed to the problem by lowering the natural resonance of the klystron mounting hardware. After replacement, there was no further evidence of the unwanted ripple voltage.

8. Special Analog Noise Test

During the AM-7, -8, and -9 system test, measurements of analog noise on the range output produced values greater than normally observed on the QA-1 through AM-6 system. Coincident with the start of this phenomenon, the single sideband modulator for the range channel in the system test equipment was replaced, the ranging facility of ECP-6 was installed, and minor maintenance was performed on test equipment spacecraft filters. In an attempt to isolate the cause of the increase in analog noise, a series of tests was conducted in accordance with the procedures established by Ryan report 51765-39.

The tests used the AM-9 system and the TE-2 RADVS system test equipment (STEA) and were initially performed with the STEA in the same configuration used for the AM-7, -8, and -9 system tests to provide a reference for further tests. The analog noise output of range channel was similar to that observed on the latest system tests. Beams 1 and 3 were then unlocked, both separately and together, and no significant change in the analog noise was detected.

A phase shifter installed in the beam 4 receive waveguide was varied over its range, and again no significant changes took place. The phase shifter was rotated in order to isolate any noise contributed by the single sideband modulators in the STEA. Although these modulators are normally adjusted for a given rejection of the unwanted sideband and the carrier, they must be adjusted at a rather high simulated return signal because of the resolution of the analyzer used in this initial setup. There is no way to measure the rejection at the lower simulated frequencies directly, although it may be indirectly measured by varying the phase shifter and noting the effect on analog noise. If there is no significant change in noise with phase shifter rotation, the rejections are adequate. For the tests described here, this measurement ensures that the noise contribution due to STEA modulator adjustment is negligible.

The switches in the waveguide run of ECP-6 were then bypassed and another series of measurements made. A slight reduction in noise level was noted but it was not sufficient to reduce the level to that observed on the QA-1 through AM-6 systems. The entire ECP-6 configuration was then bypassed by reinstallation of the waveguide run used on the earlier systems. Analog noise was immediately reduced to the previously-observed lower levels, confirming that the difference noted was due to the ECP-6 waveguide configuration. The waveguide was tested for VSWR and insertion loss and was within acceptable levels.

It is felt that the phenomenon is caused by ranging in the ECP-6 waveguide run. The noise effect is more noticeable at 50 and 200 ft than at 2000 ft. The former cases are both performed under the high deviation conditions. The range frequency resulting from deviating into a length of waveguide is a greater percentage of total range frequency than for the higher altitude cases.

Ideally, the simulated range signal consists of one single sideband signal whose range frequency is the sum of the actual range frequency of the waveguide and an additive test frequency. The actual signal contains the ideal single sideband signal together with other signals originating from various effects. At the lower simulated altitudes, the undesired signals are more likely to fall within the tracker

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bandwidth and cause noise than at the higher simulated altitudes. Complete analysis of the composite signal return would be exceedingly difficult. Additional noise present with the ECP-6 waveguide should be recognized as originating from the above discussed effects, and there will be no further analysis or investigation.

a. Test conclusions. The analog noise difference between the AM-7, -8, and -9 systems and the QA-1 through AM-6 systems is a function of the ECP-6 waveguide run. This difference has been noted on the range output, the only one affected by the ECP-6 configuration. Since the RADVS system, together with the spacecraft control system, does not fly with ECP-6 attached, noise measurements taken during system tests will be continued in the same manner as used on the AM-7, -8, and -9 system. Investigation of this analog noise difference is now terminated.

9. Quality Assurance

a. General. During this report period, significant changes in activity and improvement of the quality system were as follows:

(1) JPL quality assurance established a full-time resident quality representative at AFETR to provide better continuity of field quality requirements between programs, and quality representation for overall station activities.

(2) The permanent AFETR/HAC quality organization was staffed to a level sufficient to support a spacecraft, and the quality team following a particular spacecraft was reduced to one member.

(3) Reassignment of JPL/QA personnel to cover spacecraft activities, as previously reported, has resulted in significant improvement in spacecraft hardware integrity.

(4) HAC is in the process of revising quality procedures to achieve a standard method of operation for similar activities in its manufacturing facilities.

(5) HAC is revising and updating the company's approved materials and processes documents (MAPAR), and has released specification 291002 "Approved Materials and Process List for *Surveyor*."

b. Hardware quality.

(1) Unit consent-to-ship meetings were completed for SC-3 and were 98% complete for SC-4. The new blue book summary sheet is being used on a trial basis on selected units for SC-5 consent-to-ship meetings in an attempt to more clearly and efficiently define unit status.

(2) Additional steps have been taken to improve the integrity of electrical connectors by revising the existing quality documents and providing new "spring-loaded, plunger-type" contact socket retention-force measuring gages.

(3) Several lots of resistors from MEPCO were received and accepted without being subjected to burn-in and resistance-change tests at the supplier's facility, as required by the specification control drawing. This resulted from confusion between procurement and the supplier regarding requirements for *Surveyor* and prototype parts for another program. The problem was resolved as follows:

1) Parts manufactured by the supplier in question were withheld from issue from bonded stores.

2) Data was reviewed to isolate inadequately-tested parts, and good parts were returned to bonded stores.

3) Inadequately-tested parts were sent to the HAC components and materials group or returned to the supplier for testing to fulfill specification control drawing requirements.

4) Quality history was reviewed for all Class IV hardware using resistors of the same part and dash number as those inadequately tested.

5) Procedures were revised to include supplier data review to the extent that assurance of completeness of testing was gained. The four Class IV hardware items containing the inadequately-tested resistors were identified. Parts were replaced in two flight control electronics chassis. The material review board accepted a boost regulator and a battery charge regulator for use without resistor replacement, based on part application and maximum-expected resistance change.

6) One of the three A-22 configuration main retro-rockets (A22-7) was found by visual inspection to have a poor liner-to-insulation bond after propellant loading. Parallel investigations related to the insulation installation, liner preparation, and raw material sources have been started.

7) Due to failure in the main retro-rocket safe and arm motor, all safe and arm motors were torn down and visually inspected. Several workmanship problems were uncovered and subsequently repaired.

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8) There have been several recent cases of damage to flight hardware due to non-grounded soldering irons and lights being used during hardware rework. Review of this equipment is being made to provide necessary grounds.

C. Propulsion Engineering

1. A-21 Main Retro-Rocket Engine

Three 1300-pound A-22 configuration main retro-rocket engines were loaded to support SC-3. Postloading visual inspection of the third engine revealed a poor liner-to-insulation bond, and the engine was rejected. This defect was not observed in the other two engines.

Complete review of the steps taken in fabricating the three engines was initiated to determine the reason for the apparent differences in liner bond quality. Parallel investigations related to the insulation installation process, the liner preparation process, and the raw materials sources and quality have also been started.

To date, the only apparent difference in processing is that the reject engine insulation was cleaned with a different solvent (Chlorothane-nu) from the other two engines which were cleaned with trichlorethylene. The two solvents are considered interchangeable and have been used on that basis throughout the development and qualification programs.

There is some evidence that it is possible to procure inferior (with respect to cleaning characteristics) batches of Chlorothane-nu. Several samples were withdrawn from various drums in the Thiokol plant and allowed to evaporate from glass slides. Varying amounts of an oily residue remained on the slides. Effort is now under way at Thiokol to determine the composition, source, and effect of the residue, as well as the range which may be expected to be encountered in use. Results of this and other associated investigations will be presented in the next issue of this summary.

The loss of the engine will not be serious if it can be established that the defect is not present in the remaining engines. It is possible that this can be proven only by firing one of the engines and processing another spare. This can be accomplished within the constraints of the SC-3 launch schedule.

Safe-arm 706 failed during functional checks following flight acceptance test-level vibration. Investigation by Thiokol, HAC, and JPL revealed the cause to be a faulty electrical connection within the drive motor. Complete inspection of all safe-arm units was conducted by the drive motor manufacturer and no similar defects were found.

The safe-arm devices are being reassembled and will be retested at the prescribed FAT levels prior to use. Units designated for use on the SC-3 retro engine will be complete by December 12, and will have no impact on retro-rocket delivery schedules.

HAC is participating in the development of the case insulation installation procedure at Thiokol. This became necessary because the original insulation vendor expressed a desire to drop all aerospace business. The new procedures are being checked out on *Surveyor*-derivative, Boeing Burner II engine, and on a Thiokol (Wasatch Division) high-energy propellant program which employs *Surveyor* hardware. No *Surveyor* engines are to be processed through the insulation installation operation until Thiokol has demonstrated its capability to the full satisfaction of HAC.

2. Vernier Propulsion System

a. Systems analysis. Because of the mission B failure, an extensive investigation was made by JPL and HAC teams of the failure of thrust chamber assembly 3 ignition during the midcourse maneuver. An extensive series of SC-2 simulation tests was conducted. The following conclusions were reached from a detailed review of the SC-2 data and comparison with simulation test results:

(1) Engine 3 had approximately full fuel flow during three of the attempted firings and probably during all firing attempts.

(2) Engine 3 had little, if any, oxidizer flow during midcourse.

(3) In post-midcourse firing attempts, engine 3 had less than the commanded oxidizer flow.

(4) Engine 3 did not fire during any of the attempted firings.

(5) During two firings, engine 1 apparently shut off 1-2 sec later than commanded.

(6) Engine 2 temperature rise was excessive both in the 2-sec burns after burns 2 and 3 and in the 21.5-sec burn.

(7) Helium tank pressure displayed unexplained fluctuations (suspected bad transducer).

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(8) Change in unregulated bus current varied from 1.4 to 2.4 amp when engines were turned on. Based on the resistance of the solenoid valves, current change should be 1.7 amp.

The failure review board is currently preparing a report on its investigation.

b. Helium tank and valve assemblies. Type approval tests for the A-21E configuration have been completed at Wyle Laboratories. The test data have been compiled and a report submitted to HAC by Wyle. All phases of the type approval tests were satisfactorily completed.

c. Helium regulators. Water expulsion tests are being performed on a regulator at a remote test site in an attempt to simulate the sequence of events that occurred during mission B. Dynamic behavior of the regulator is being investigated. Test results to date indicate that the SC-2 helium regulator performed satisfactorily.

d. Check and relief valves. All control items for A-21E have completed type approval tests with no out-of-specification discrepancies noted in their performance. The final type approval test report is being prepared and will be completed early in the next reporting period. Valves for SC-7 were received and have successfully completed flight acceptance test.

e. Fill and vent valves. Cycling tests were performed on three fill and vent valves to determine whether contamination could develop in the valves as a result of an excessive number of opening and closing cycles. Each valve was cycled 100 times and, following every 10 cycles, a millipore filtration and contamination count was taken from a sample of liquid from the valves. The tests showed that contaminants are not generated to a level detrimental to the *Surveyor* vernier propulsion system after many opening and closing cycles of the fill and vent valves.

f. Propellant tanks. Inspection of several propellant tanks revealed a concavity of the weld bead which, in the worst case, was approximately 0.003 in. below the parent material. A special test, consisting of 300 pressure cycles from zero to the maximum operating pressure, was conducted on propellant tank 147 prior to a standard one-in-ten burst test. The propellant tank successfully passed the cycle and burst test. Normalization of the actual burst pressure met the specification requirements. The remaining tanks will be delivered early in the next report period.

g. Metallized bladders. Gas-permeation tests on metallized bladders were completed. Tests with oxidizer at the 240-psi helium pad pressure level showed that saturation of the oxidizer occurs in approximately 10 days for a new bladder. If the bladder has undergone flight acceptance and type approval test vibration levels, saturation occurs in one day. Tests at 220-psi helium pad pressure using MMH showed that permeation rates are lower than for the oxidizer and that saturation of a bladder which has experienced flight acceptance and type approval test vibration occurs within 13 days.

A comparison test performed at 800 psi, using the oxidizer in both an all-Teflon bladder and a metallized bladder revealed that the all-Teflon bladder allowed 75% saturation in 16 hr compared to 75% saturation in 66 hr for a metallized bladder having undergone flight acceptance and type approval test vibration. The permeation tests demonstrated that, although the metallized bladder achieves significantly-reduced permeation rates as compared to the all-Teflon bladder, the oxidizer would be completely saturated at pad pressure before launch and approximately 75% saturated before terminal descent.

Three-month storage tests of metallized bladders in both reference propellants were completed, and these bladders are currently undergoing expulsion tests. Fifteen expulsion tests have been completed on bladders which have been through either flight acceptance and type approval test vibration or various storage periods in propellants.

Based on the results of both permeation and expulsion tests, it was decided not to incorporate metallized bladders in the current *Surveyor* series.

3. Thrust Chamber Assembly

The program for *Surveyor* vernier engine (RMD Model No. TD-339) evaluation continued with 60 engine firing tests, and completion of the improved fuel poppet material investigation. In addition, there were many solvent and single propellant flow tests which were part of the SC-2 failure analysis effort.

a. Teflon poppet evaluation. As reported in SPS 37-39, p. 64, a poppet with a plain Teflon seat was assembled in a shutoff valve for evaluation under lunar survival temperatures (+300 to −300°F). Tests DY 105–109 indicated that the Teflon seat would seal successfully following 90"on-off"cycles in the 0–100°F temperature range and after storage with fuel at 300°F for five days. However, leaks developed upon repeated attempts to cycle the valve to cold temperatures (Test C-260) and the valve (S/N 303) was removed from the test.

A second valve (S/N 333) was equipped with a poppet made from a filled Teflon-TFE composition produced by

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Ethylene New England Products Corporation. This material, containing 18% by weight of silica filler and having a Shore-D hardness of 54, survived 75 on-off cycles in the 0–100°F temperature range; but it developed excessive leakage (420 cc helium/hr) following the 90th cycle (tests DY 110–116). Therefore, testing of this particular glass-filled Teflon formulation was discontinued.

Analysis of the Teflon material failure led to the conclusion that excessive plastic deformation of the fuel poppet seat had taken place during high-temperature storage, leading to leakage during the transition to low temperatures. It was decided that certain mechanical modifications should be made to the valve, and that other grades of glass-filled Teflon should be considered. Valve modifications included:

(1) The addition of a spring guide to minimize the possibility of spring cocking and resultant side loads on the fuel poppet.

(2) Machining to improve the seat surface finish.

(3) Tighter controls on metal seat dimensions to minimize bearing stresses.

At the same time, a glass fiber-filled TFE Teflon formulation identified as Fluorosint (produced by Polymer Corporation) was machined into poppet form for the two test valves. This material has a nominal filler loading of 54.5% by weight and a Shore-D hardness of 68 (approximately equivalent to that of Nylon-101). Its high compressive strength, together with a low coefficient of expansion, made it appear worthwhile to continue the search for a material which can survive lunar temperature extremes.

Each valve was subjected to 90 on-off cycles at 100°F (tests F-23 and 24), followed by an abbreviated lunar-survival storage simulation (tests F-25 and 26). Both valves survived 6 days of storage at 300°F, following by cycling to −300°F, storage at −300°F for 1.5 days, and return to ambient temperature with no fuel leakage by the poppet. Maximum leakage by the poppets after testing was 48 cc helium/hour. Examination of the glass-filled Teflon poppets following test showed no evidence of material degradation. The metal-seat impression on the glass-filled Teflon was approximately 0.009-in. deep.

It is possible that improvement in survivability resulted from the two valve modifications, but the superior hardness of Fluorosint appears to be a more likely explanation. Consequently, the Fluorosint used in the modified valve design is believed to be a satisfactory Nylon-101 poppet material substitute under lunar temperature extremes.

b. Off-design lunar tests. Nine tests (DD 131–137, 140, and 142) were performed in August to determine transient and steady-state dynamic performance of the engine under conditions anticipated for an SC-2 lunar liftoff and translational maneuver. Propellants, engine, and line-temperature conditioning prior to each firing was based on SC-1 data and covered a range from ambient to 230°F. All firing tests were conducted with helium-saturated propellants. Two firings (DD 135 and 136) were performed at a minimum thrust level—the first part of the firing profile involving a pulse train between −30 and −78 Δma. All other firings were performed according to a 50-lb thrust control tape with a dynamic profile of 49-sec duration. Start and shutdown transient data at both minimum and mid-thrust levels were obtained. All analog tapes were transmitted to the HAC flight control group for evaluation of the dynamic response of the engine under lunar simulated temperatures. It was concluded by HAC that the frequency response of the engine at simulated lunar off-design temperatures was not much different from its response at design temperatures. Start and shutdown transients for a firing at the minimum thrust level with high temperature propellants, however, are of considerably longer duration than for a comparable firing with design temperature propellants.

c. Mission-B lunar survival tests. Three tests (DD 143–145) were performed in early September to determine the ability of the engine to survive lunar-liftoff and translational-maneuver firings for mission B postlanding operations. Of major concern was the ability of the shutoff valve assembly Nylon-101 fuel poppet to survive high temperatures associated with postlanding firings. A sea-level engine (S/N 531) was stored under vacuum with temperature-conditioned (220–250°F) and helium-saturated propellants under full tank pressures for three days and fired for a short period once every 24 hours. The engine performed the simulated mission successfully, although start and shutdown transients were considerably longer than normal. Post-test examination of the Nylon-101 poppet disclosed an intact sealing surface, yet evidence of cracking and deterioration from its contact with the high-temperature fuel.

d. Lunar dust firings. As reported in SPS 37-40, an altitude engine was oriented to fire vertically down into a container filled with a simulated lunar soil during two tests. Results indicated sufficient energy in the exhaust gases to transport and raise to significant heights an appreciable quantity of soil.

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In August, two attempts (tests DD 138 and 139) were made to establish a vacuum and fire an engine over fine loose silt as a part of a program to determine the erosion pattern caused by vernier engine exhaust gases over a simulated lunar surface. In both tests, during evacuation of the test chamber with a mechanical pump, boiling of the soil took place at a pressure of about 0.10 psia. As a result, the engines were not fired. Subsequently, the soil was placed in an oven and heated to 275°F for about 5 hours to drive off moisture and other volatiles. The soil was then held under vacuum for about a week until additional tests could be scheduled. Tests DD 146, 148, and 149 were successfully conducted in late September with engines firing into three different grades of soil in a vacuum of about 0.01 psia. Erosion data, soil pressure measurements, and high-speed movies were taken. The movies vividly illustrated the movement of loosely-packed coarse soil away from the central zone during engine firing and the violent eruption of loosely-packed fine soil upon engine shutdown.

Soil pressure gages indicated peak-pressure values occurred shortly after full chamber pressure developed; thereafter, soil pressures decayed rapidly to end of test. Maximum soil pressure was recorded by a gage centered under the engine, and highest pressures developed under the loose/fine material. Data taken in test DD 149 during vacuum chamber pumpdown indicated an amplification of pressure existed for a gage buried under the soil when compared with readings from a gage mounted on the vacuum chamber wall. Since the difference in measurements was considerably larger than pressure resulting from the weight of soil over the buried gage, additional testing was required to interpret these pressure readings.

e. SC-2 failure analyses supporting tests. A series of tests was conducted at the Edwards Test Station in support of analyses of the SC-2 failure during October, November, and December. Purpose was to gain information which would provide a basis for interpreting telemetered flight data. The tests consisted of:

(1) Single-propellant and solvent-flow simulations of the SC-2 leg 3 failure mode.

(2) Engine-firing simulations of flight conditions for all three SC-2 engines.

(3) Engine firings simulating conditions for SC-1 engines.

Test conditions were chosen to simulate many burns on SC-2 engines 1 and 2 as well as possible failure modes of engine 3, and tests were repeated to determine the reproducibility and validity of data.

Since engine-case and oxidizer-line temperatures were the major telemetered indicators during the 41 burns of SC-2, particular attention was given to these measurements in the subject ground test program. Other telemetered data, which were directly related to engine behavior, were propellant tank temperature at the time of a particular burn and electrical input to the solenoid valves. For a majority of the ground tests, propellants were preconditioned to whatever temperatures were recorded from flight data for each set of conditions to be simulated. Flight lines and engine brackets (for 1, 2, or 3) were used in conjunction with proper thermal isolation from the test stand structure to provide representative thermal paths and more precise ground test temperature data.

Since it was important to simulate heat transfer characteristics of particular flight engines, the two altitude engines (S/N 535 and 538) which provided the bases for thermal comparison were acceptance-tested to determine their heat transfer characteristics. Each engine tested, therefore, depended on which flight the engine was to be simulated.

The SC-2 failure analysis program was initiated with non-firing flow tests (DD 150–155) using S/N 09 engine in which oxidizer and fuel were flowed separately through the flight lines and engine to examine the leg 3 engine failure mode. Flight data during attempted firings had indicated marked decreases in the engine outer case temperature. The test objective, therefore, was to determine whether the change in engine temperature could be associated with the flow of one of the propellants. The engine was temperature-conditioned either hotter or colder than the propellant temperature in simulation of spacecraft conditions for the particular burn period. In tests DD 150, 152, and 155 (with the engine conditioned hotter than the propellants) about the same amount of cooling was developed with oxidizer flow as with fuel flow. In tests DD 153 and 154 (with the engine conditioned colder than the particular propellant to be flowed), the engine warmed up to about the same temperature as existed in the propellant supply circuit and then cooled off below its original temperature.

These tests were followed by solvent flow tests, (C-78 A-P and C-79 A-I) in which the effect of "intermittent current pulses to the solenoid valve" on engine flow characteristics was to be determined. It was suspected that the current to the three solenoid valves may have been

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characterized by pulses with an amplitude of about 0.5 amp superimposed on a steady dc level of about 1.25 amp. The degree to which such a wave form could affect the opening characteristics of the shutoff valve assembly was to be tested. The second series of tests (C-79 A-I) was conducted with a flight-type power supply system which produced a pulsing current input to the shutoff valve assembly at various duty cycles and pulse repetition rates. The flow tests showed that it was possible to produce no flow, fuel flow only, fuel and partial oxidizer flow, or full fuel and oxidizer flow. The particular flow regime depended on the pulse rate and duty cycle.

Firing tests were then made with the S/N 09 engine to determine whether the SC-2 engines of legs 1 and 2 had performed properly in flight. As part of the test series, thermocouples were welded to the engine outer case to provide a means of assessing the accuracy of the engine case flight sensor (a resistance thermometer which is resin-bonded to the case). Tests were made with pulsing as well as steady shutoff-valve current, and they indicated that the spot-welded thermocouples were much more responsive to sudden temperature changes than the flight sensor. Tests DD 163 through 165 indicated that with a pulsing signal (44–46% duty cycle) to the solenoid valve, both flows could be forced into a pulsing mode. The fuel flow was affected to a larger degree than the oxidizer flow. In one test, fuel flow was completely interrupted during the off-cycle. Data also indicated that heat rejection rates can be nearly doubled in a pulsing mode.

To confirm suspected low-heat transfer rates of the S/N 09 engine, test DD 167 was performed in simulation of conditions on leg 3 of SC-1 during the first 38 sec of terminal descent. The low heat rejection rate of the S/N 09 engine was indicated by comparing flight data with ground test data.

A series of solvent flow tests (C-81) was performed with four representative sea-level engines to determine the range of helium-supply pressures required to open and close the two poppets (oxidizer and fuel) of the shutoff valve at several throttle-valve settings with nominal tank pressures of 720 psia and various helium-actuation pressure levels. All four engines reacted similarly. With increasing helium-actuation pressure, the fuel poppet opened at about 150 psi, followed by the oxidizer poppet when the pressure reached 450 psi. With decreasing pressure, the oxidizer poppet closed between 250–150 psi, followed by the fuel poppet at 80–45-psi actuation pressure level.

Since tests of the S/N 09 engine had indicated that its thermal behavior was quite different from that of spacecraft engines, two other flight-type engines (S/N 535 and S/N 538) were introduced into the program.

After tests with engine S/N 538, data indicated the thermal mass of the mount had an appreciable influence on engine thermal behavior. As a result of insulation additions at selected points, ground test data for the S/N 538 engine more accurately simulated flight data. Thus, it was demonstrated that this engine, when properly thermally isolated, could be reasonably used in additional tests to explore SC-2 failure modes.

To further explore the SC-2 leg 3 failure mode and gain additional information on the cooling characteristics of an engine and flight line with one propellant flowing in the system, a series of flow tests (DD 177 A-E and DD 177 H and I) were performed with the S/N 538 engine mounted on improved and thermally-isolated bracketry. Oxidizer and fuel were alternately flowed at various rates through the flight lines and out the engine bleed ports; and, in each test, the propellant supply was conditioned about 10°F colder than the engine in simulation of spacecraft temperature conditions to determine thermal behavior and time to reach thermal equilibrium with the propellant supply.

The S/N 535 engine was next installed in DD stand with a shutoff valve modified by addition of a retaining screw to restrict the movement of the oxidizer poppet by predetermined amounts. The engine thrust to be simulated was approximately two lb, and oxidizer line temperature changes were about 3°F. The test program consisted of a series of off-mixture ratio start tests (DD 178–183), in simulation of the possible failure mode during the second burn of the SC-2 leg 3 engine. On the sixth firing attempt, the engine developed a minimum level of thrust (13 lb peak at the start, falling to 4 lb at the end of the 2-sec firing period) and operated at a mixture ratio of 0.086. Oxidizer line temperature changes were about twice the value being simulated.

The program continued with two firing tests, each lasting 21.5 sec, with S/N 535 and 538 engines in simulation of the fortieth burn of SC-2 leg 1 and leg 2 engines. Engine S/N 535 simulated the leg 2 engine in test DD 184; engine S/N 538, the leg 1 engine in test DD 185. Firings were made with an HAC-supplied tape which simulated spacecraft command signals. The engine and propellants were preconditioned to the temperatures indicated by telemetered spacecraft data at the time of final burn. Again, the

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thermal behavior of the engine during and following the firing was the criterion for determining whether the leg 1 and 2 engines operated properly. The engine case temperature data from test DD 184 indicated poor simulation, in that the temperature in the ground test was considerably lower than that recorded in flight. At a later date (16 December), a third test (DD 202) of the S/N 535 engine (in simulation of the fortieth burn of the SC-2 leg 2 engine) was performed. This test was a rerun of earlier tests (DD 174 and 184) with improved thermal simulation. Engine case temperature data in test DD 202 were higher than that recorded in test DD 184, but still were not as high as that recorded in flight. This again illustrated the difficulty of closely simulating flight thermal conditions in a ground test.

The S/N 538 engine case temperature data from test DD 185 exhibited close simulation with flight data during the engine firing period.

The first portion of the SC-2 failure analyses supporting tests was concluded on 10 November with solvent flow tests (C 103) through two sea-level engines. Of particular interest in these tests was the sequence with which oxidizer and fuel flows develop in the 400–500-psi pressure range, since this is the range in which the fuel regulator and oxidizer poppet are expected to actuate fully open. Both shutoff-valve and fuel-regulator operating characteristics were well-defined.

Tests DD 189–202, completed on 16 December, brought the current testing effort in support of SC-2 failure analyses to an end. Of special note is test DD 189, related to conditions existent during burn No. 27, since the firing profile simulated that of leg No. 1 engine which may have operated for 2.3 sec at mid-thrust level following the signal to shut down. Engine case temperature-time history for the firing and soak period was in good agreement with flight data, but oxidizer-line temperature changes were only in fair agreement.

D. Electrical Power System

1. Main Battery

Main batteries 67 and 68 developed shorts in cell 12. Battery 67 was examined at Electric Storage Battery Company (ESB) and battery 68 was examined at HAC. In both batteries, the point of degradation was between the polystyrene support strut and the uncoined positive plate area. Fig. 1 shows the short at the coin area between the strut and positive-plate uncoined area of battery 68. Fig. 2 shows a normal plate from the same battery. Analysis is in progress to determine the failure mechanism and reduce or eliminate the possibility of a similar failure occurring on future batteries. So far, only two in approximately 20,000 plates have exhibited this anomaly.

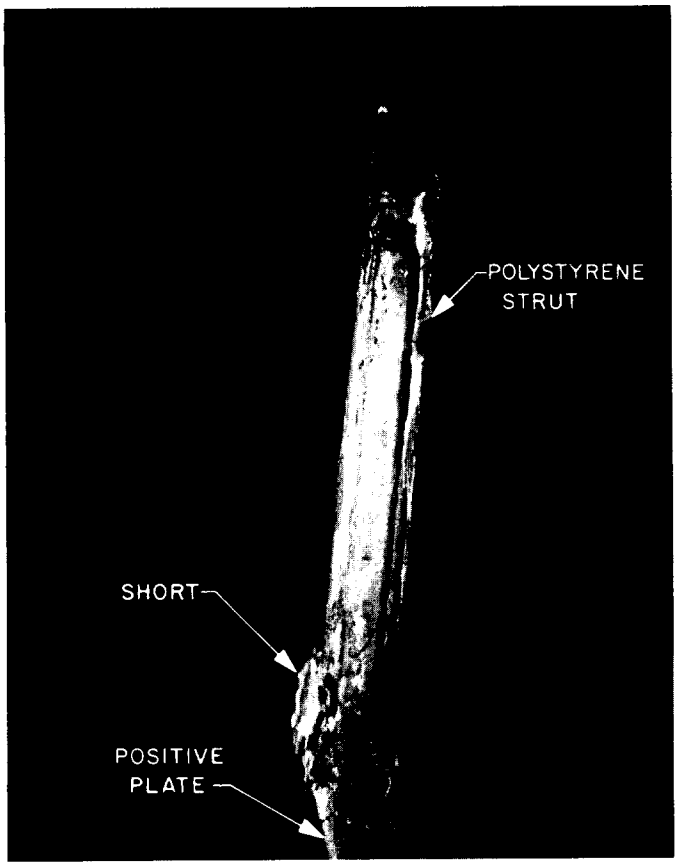

Fig. 1. Short in coin area between strut and positive plate

During dissection of battery 68, crystalline growths were observed on the positive plate (Figs. 3 and 4). These fairly hard crystals appeared to be denting the negative plate; none were seen to penetrate the separator. The crystals are probably generated by the attack of silver oxide (charged state of the positive plate) on the cellophane separator.

Considerable effort is being expended in the battery area as a result of the A-21E program. With the elimination of the auxiliary battery from SC-5 and subsequent spacecraft, the two-battery operation of *Surveyor* during the heavy loads of terminal descent has been eliminated and the main battery will be required to supply increased current during that period. Expedited testing is being

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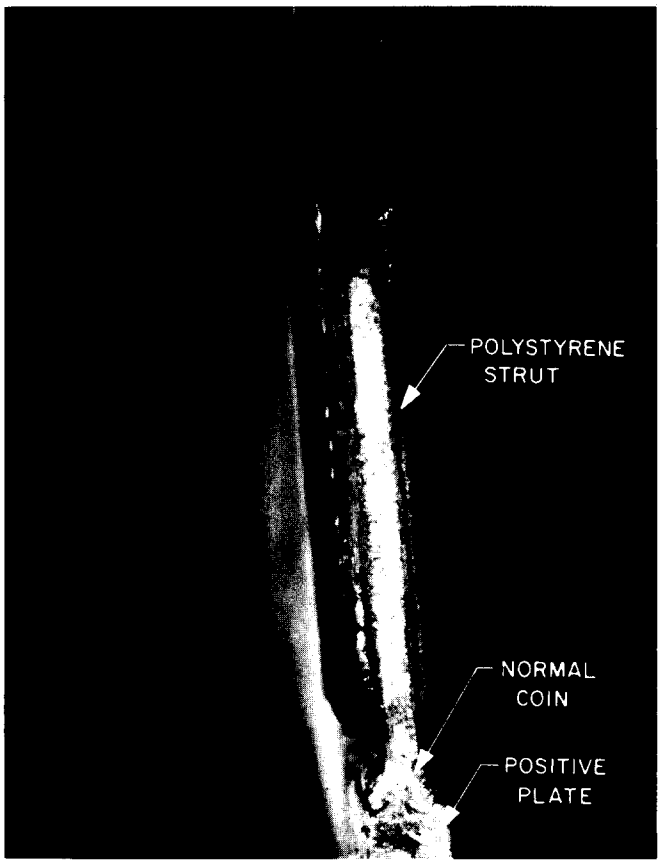

Fig. 2. Normal coin area between strut and positive plate

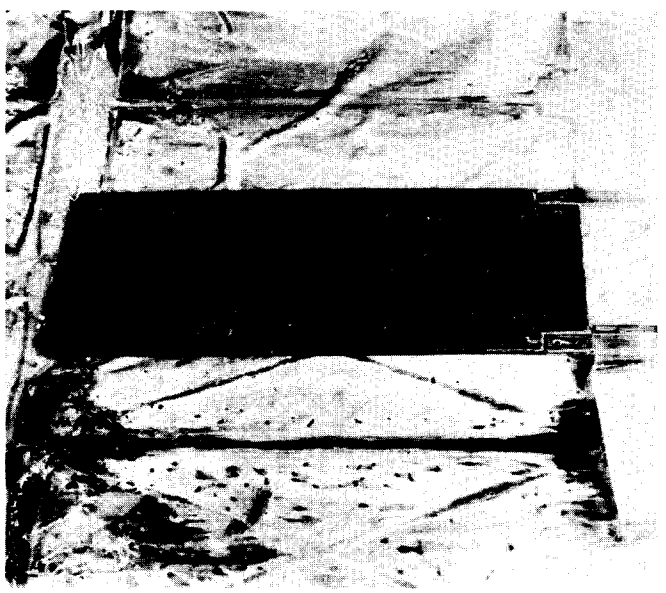

Fig. 3. Dents in negative plate because of crystal growth

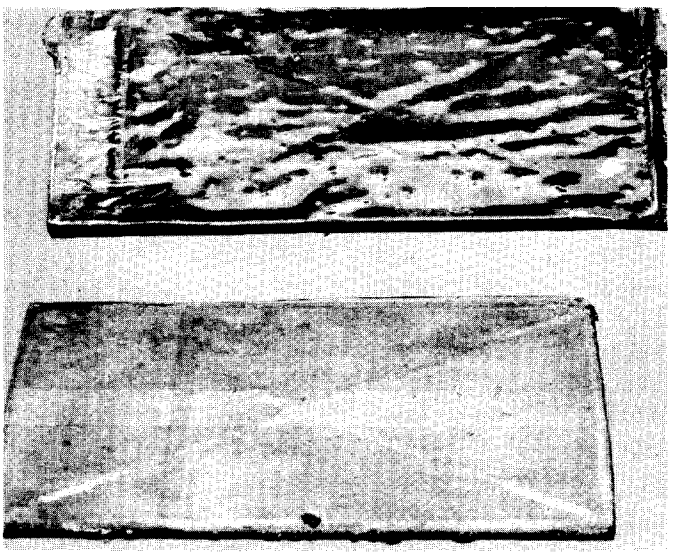

Fig. 4. Crystals on positive plate

conducted to establish battery capability under the operating conditions of a single battery system.

Construction of a calorimeter for studying thermodynamic response of the main battery under varying load conditions has been completed. Fig. 5 shows an overall

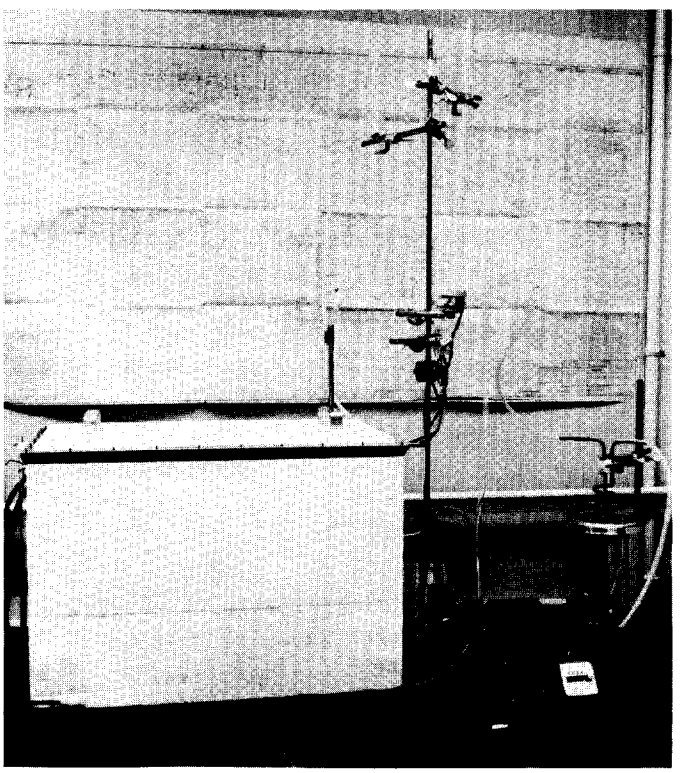

Fig. 5. Overall view of calorimeter

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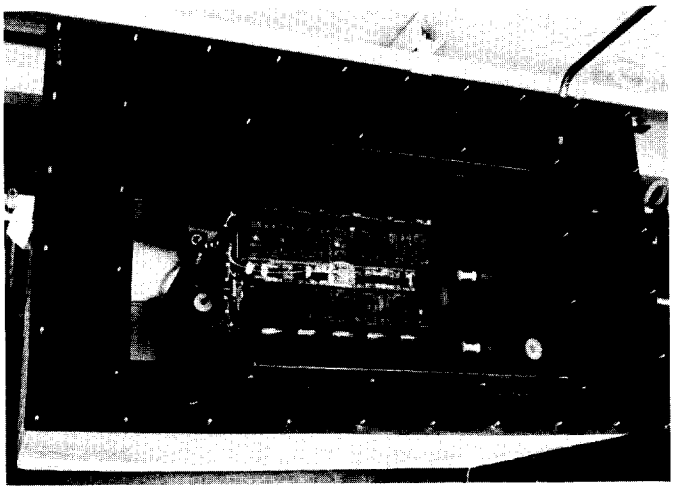

Fig. 6. Calorimeter with top removed and battery in test position

view of the calorimeter setup. Fig. 6 shows the calorimeter interior with cover removed and main battery in position for testing. Initial calibration and checkout is in progress.

Batteries 97 and 98 are being built with thermocouples located on four negative plates, on the outside of four cell jars, and on the ends of the monobloc assembly. These batteries will be used to study the internal heat generation patterns of the main battery.

Batteries used for solar thermal vacuum and mission sequence electromagnetic-interference tests during the reporting period successfully met the system requirements.

Anticipated high-rate battery discharge for SC-5, SC-6 and SC-7 was simulated using main batteries 52 and 58. The batteries were discharged for 10 hr at a 7-amp rate (70 amp-hr withdrawn). Remaining battery capacity was approximately 100 amp-hr for each battery. The batteries were then discharged at a 50-amp rate with random 20-amp pulses for 50-μsec duration applied. All testing was at room ambient temperature and pressure.

E. Scientific Payload

1. Alpha Scattering Experiment Subsystem

a. Subsystem design and integration. The subsystem design and integration effort for the alpha scattering experiment during this reporting period has included:

(1) Design and development of HAC controlled items.

(2) Preliminary release of the detail subsystem performance/design and test requirements specification.

(3) Release of the subsystem wiring diagram.

(4) Product development coordination.

(5) Special analytical studies.

TV viewability. A TV viewability study of the sensor head deployment area was completed during the reporting period. Results indicate that the instrument sensor head obstructs visibility of the deployment area. A subsidiary viewing mirror is therefore being fitted to the spacecraft.

Operational sequences. The definition of the operational sequence of the alpha scattering experiment is continuing, and preliminary versions have been generated. Included are sequences for checkout of the instrument at HAC during payload systems tests and solar thermal vacuum (STV) tests, at AFETR before launch, and for in-transit operation and lunar operations (stowed, background, and surface counts). The in-transit operation of the experiment is a potential new requirement of the program and is being investigated for the feasibility of its implementation.

Subsystem testing. Test plans were prepared for a special vibration response test of the alpha scattering instrument deployment mechanism (ASIDM) using the S-9 test vehicle as the vibration test support fixture. The special vibration test of S-9 will explore ASIDM response to spacecraft flight acceptance test levels. The above tests are scheduled during January, 1967.

Payload systems test laboratory. The alpha scattering experiment subsystems test laboratory was reactivated during the reporting period. Scientific payload system test racks and the payload systems laboratory spaceframe model (M-13) were assembled. Modification of the above equipment is in process to reflect the latest design changes in the alpha scattering subsystem. Completion of the test equipment is scheduled to coincide with the delivery of hardware. At present, a boost regulator and low data rate auxiliary unit have been added to the payload systems laboratory basic bus complement. The central signal processor, engineering mechanism auxiliary, and boost regulator units are scheduled for addition before tests begin.

b. Alpha scattering instrument auxiliary (ASIA). The ASIA provides command decoding, data signal processing, power management, and squib firing control for the experiment. Command decoding is accomplished by a subsystem decoder which generates the 17 command outputs required for subsystem operation from input commands received from the basic bus central command

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decoder. Two data channels, one for the alpha particle data and another for the proton particle data, are provided by two frequency modulated subcarrier oscillators. Power management functions include a 22-v switch for the sensor heater, a 29-v switch for the instrument power, and a 29-v switch for the auxiliary power requirements. Squib control is provided by two power switches, one for the deployment to the background count position and the other for the deployment to the lunar surface count position.

During the reporting period the circuit design and product design of the auxiliary were completed on schedule. Final design review for these efforts was conducted, with no significant changes being required.

Fabrication of the units continued on schedule with all modules, including spares, welded and foamed. All parts are on order, and no shortages are expected.

Preparations are now being made for unit flight acceptance test. Completed thus far are the preliminary test plan and the design of modifications to the test equipment.

c. Flight electrical equivalent. The flight electrical equivalent is being fabricated to terminate basic bus signal loads when the experiment equipment is not installed and, if necessary, to fly in place of the experiment equipment; this item has been designed and the drawings released. Fabrication of the first model is proceeding as scheduled.

d. Alpha scattering instrument deployment mechanism (ASIDM). The outline and mounting drawing, contour cable wiring schematic, and all fabrication drawings were released for manufacture during the reporting period, and fabrication of the component parts was completed. Component assembly of the first two units has begun and no schedule problems are expected.

As part of the development program, a design review of the deployment mechanism was conducted and included the results of the vibration tests discussed below.

e. Vibration tests. Appraisal of engineering evaluation vibration testing data for the ASIDM resulted in a special vibration response test being scheduled and preparations initiated to use the upgraded S-9 test vehicle as a test support fixture.

Analysis of the engineering evaluation vibration tests indicated that relatively high amplification factors could occur during vibration of the ASIDM. Therefore, a parallel design was started to reduce the vibration amplification factor in the present ASIDM configuration by incorporating damping in the sensor support structure. A redesign of the support strut was completed and is in fabrication. The modified support strut will be available for the upcoming S-9 vibration tests.

f. Deployment tests. Engineering evaluation deployment tests were conducted at various lunar surface slope angles varying from 0–35 deg slope inclination. Satisfactory deployment was achieved for each of the inclination angles.

g. Analytical studies. An analytical study of ASIDM release and deployment phases was initiated. The equations characterizing the position of the sensor head (when the support cord is taut) are being studied as a part of this analysis. These equations consider the instantaneous forces and moments applied by the cord. Parametric curves obtained from the equations will indicate, as a function of the length of the cord and the inclination of the spacecraft with respect to the lunar surface, the difference between the forces acting on the sensor head and the force required to overcome the frictional resistance and to release the sensor head from the support ring. Comparison of these forces will indicate both the margin of safety available for positive deployment of the sensor head and possible parametric values that would prohibit the deployment of the sensor head.

2. Soil Mechanics/Surface Sampler (SM/SS) Experiment Subsystem

a. Introduction. A SM/SS subsystem, consisting of a mechanism, auxiliary electronics, thermal compartment installation substructure, and wiring harnesses, is being developed for incorporation on SC-3 and SC-4. The mechanism, controlled by the auxiliary electronics which receives and decodes ground commands, will perform picking, digging, scraping, and trenching operations on the lunar surface. The objective is to design, develop, and flight-acceptance test class III flight units for delivery to AFETR for installation on SC-3 and SC-4 during spacecraft checkout.

b. SM/SS subsystem integration. The subsystem performance/design and test requirements specification has been upgraded to include all SM/SS spacecraft interface requirements, performance and design requirements for all elements of the subsystem, engineering test and evaluation, flight acceptance test, and spacecraft integration test requirements.

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Installation substructure and thermal-mass mockups of he mechanism and auxiliary were fabricated and installed on SC-3 and SC-4. STV test of SC-3 was performed with these units installed. The auxiliary mockup was replaced with the auxiliary flight unit for the vibration test of SC-3. The auxiliary successfully passed this test, and vibration data obtained on the mechanism mockup was used to generate test levels for the vibration test of the mechanism flight unit.

The flight wiring harnesses that interconnect the auxiliary with the approach camera TV-4 wiring harness and the auxiliary with the mechanism were fabricated.

c. SM/SS auxiliary. The auxiliary consists of electronics and a thermal compartment. Final design review of the auxiliary circuits was held, and the engineering breadboard of the auxiliary electronics has been successfully tested over an extended temperature range. Two auxiliaries were fabricated and successfully tested at the unit level. A third (spare) unit is being fabricated. Two testers, one for unit testing and one for subsystem testing, were fabricated and checked out.

d. SM/SS mechanism. All drawings and the supporting thermal and preliminary stress analyses were completed. Thermal vacuum testing of prototype motors was successfully completed. Two sets of HAC-modified motors and one set of AiResearch-modified motors were acceptance tested.

The first mechanism flight unit was fabricated, assembled, and functionally tested under laboratory ambient conditions. Vibration testing of this unit, using levels resulting from the responses of the mockup on SC-3, was successfully completed. Fabrication of the second unit was completed and assembly begun.

F. Engineering Mechanics

1. Structures

a. Shock absorbers. Type-approval testing, including man-rated burst testing, of the new-generation shock absorber 238927 was completed and a report submitted by the National Water Lift (NWL) Company. All of the type-approval test objectives were successfully achieved. Failure mode vibration and drop tests were completed and have demonstrated that the unit surpasses the design failure levels.

b. Touchdown dynamics. The A-21E phase of the S-15 stability drop test program is complete and results are documented in S-15/A-21E stability drop tests, SSD 68250R. Six two-dimensional drop tests were made on a 15-deg slope with 2-in. cubic abutments positioned in footpad impact areas. Test results indicate that adequate vehicle stability is attained with shock absorbers 238927. Demonstrated stability is in excess of analytically predicted capability and design requirements.

c. Dynamic analysis. A deployment test is conducted on each landing gear to assume that there is sufficient spring torque to overcome any friction and deploy the legs. These tests are conducted several times during the preparation of the spacecraft and are very similar to the tests performed on the omnidirectional antennas. The test consists of measuring the force necessary to initiate deployment of the stowed landing gear under earth gravity. If this force is above a certain value, it indicates that there is excessive friction in the landing gear mechanism, and steps are taken to remedy the problem. Work has recently been completed to more precisely define the limiting acceptable values for the measured deployment force as a function of each leg configuration.

The upper value for this deployment force was determined using a constant-value friction model. However, the allowable maximum value has been reduced to provide a margin of safety for deployment of the landing gear. This reduction of the upper limit accounted for the following factors:

(1) The spacecraft roll axis may not be vertical, as has been assumed in the calculations.

(2) The leg should arrive at the deployed position with some residual velocity to ensure deployment lock.

(3) The friction model assumed may be slightly incorrect.

The lower limit for this force corresponds to a situation where there is no friction, and a value below this would merely indicate that the test was not conducted properly.

One assumption made in determining the test force envelope was that the vehicle roll axis is vertical. The magnitude of the test force is quite sensitive to deviations of the vehicle attitude—e.g., a deviation of ½ deg from the vertical can cause a change in the test force of as much as 1¼ oz. It has been recommended that when the deployment test is being conducted the vehicle roll axis be vertical to within ½ deg or less.

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d. Dynamic tests. The S-9 test vehicle was returned to HAC following completion of a modal survey and controls vibration test program by JPL. A structural upgrade is currently being performed on the S-9 vehicle. This work includes removal of the auxiliary battery, the TV approach camera, and the substructure for both units. An alpha-scattering mechanism of flight quality will be installed in the vicinity of the removed units. Upon completion of the upgrade, the S-9 vehicle will be vibrated to A levels (VFSW only), and the response of the alpha-scattering sensor head measured during exposure to the flight acceptance test environment.

Following the sensor head evaluation, the S-9 model will again be upgraded to include major structural and unit changes incorporated in the SC-5 vehicle. The upgrade vehicle with a 293275 model A/SPP will be subjected to C level vibration tests and worst case A/SPP lunar landing shock tests.

e. Test evaluation of shipping container breather valves. Stratotech performed tests for determining relief pressures on two-way breather valves on three of their specimens. The test was witnessed by members of the cognizant engineering and design groups and a member of the quality control group. An investigation was conducted on another valve to determine which, if any, of the various valve components were made of plastic material. This test program was initiated because of the problems encountered on the 261955 antenna/solar panel positioner container during SC-2 transportation to AFETR.

The test for determining breather valve relief pressures was conducted on the equipment as shown in Fig. 7. Fig. 8 defines the "in" and "out" directions for pressure application. The test was conducted at room temperature.

The two-way breather valves were mounted in a leak-proof air chamber so that air pressure could be applied to the valve in the "in" and "out" directions, alternately. This required mounting and applying air pressure so that the pressure was in the "out" direction, dismounting the valve, turning it 180 deg, remounting, and applying the pressure so that it acted on the valve in the "in" direction.

The two-way breather valve pressures were monitored by a manometer from which readings were taken in in./Hg and converted to psig pressure.

One valve (622P5V5) which had been previously tested for relief pressure readings was disassembled, and each

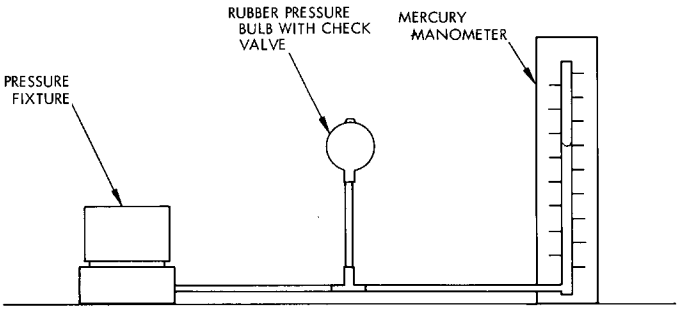

Fig. 7. General test layout

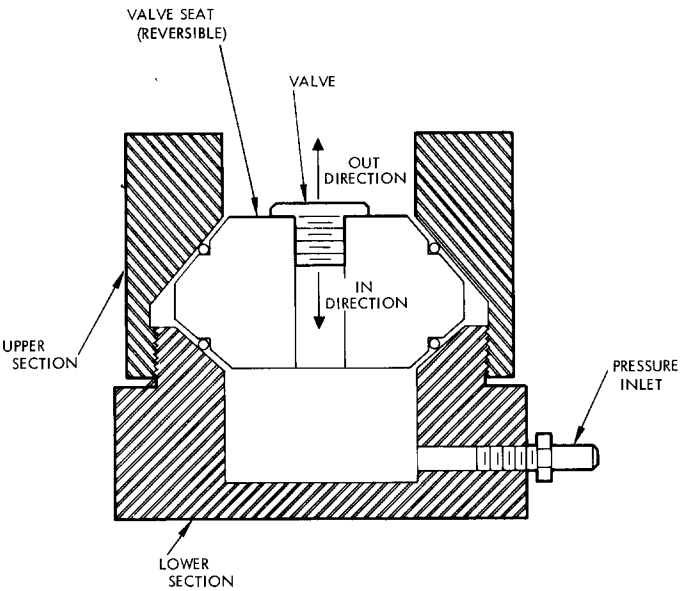

Fig. 8. Detail of pressure fixture for valve testing

valve component was investigated to determine its material type; none of the components was made of plastic material.

All relief pressure readings obtained are favorable with respect to the required tolerance band specified. These test results are but a part of the empirical data which will be accumulated from similar tests on two-way breather type valves. They will serve as a guide to set up a quality assurance program calling for a periodic inspection of the valve pressure relief tolerances which can be affected by humidity and aging. These test data will also be helpful in establishing valve shelf life criteria.

f. Stress analysis of soil mechanics/surface sampler (SM/SS). The SM/SS originally designated for the A-24 scientific payload *Surveyor* has been modified and will be flown on SC-3. The extension-retraction mechanism for

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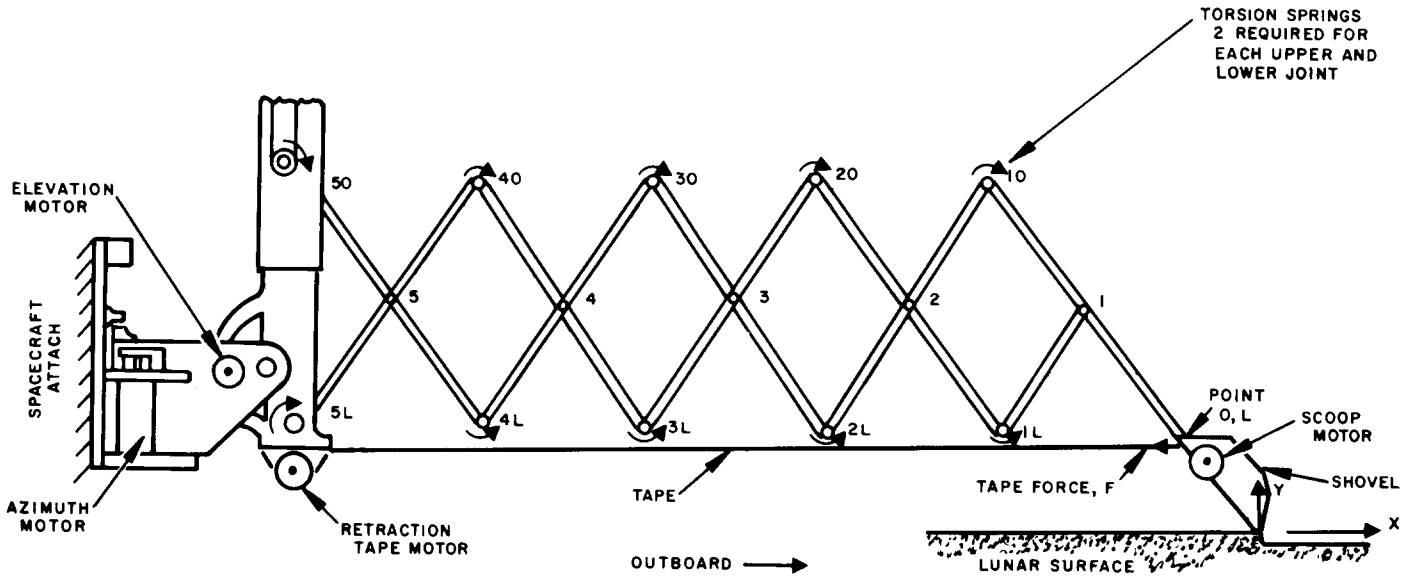

Fig. 9. Soil mechanics/surface sampler—extended position

locating the scoop-shovel on the lunar terrain is a lightweight scissor mechanism (lazy tongs). This shovel is extended on the lunar surface by springs in the linkage joints, pushing the shovel out, while the tape exerts a small force in the opposite direction. After the shovel strikes the surface, the mechanism is retracted by means of a motor drive which exerts a tensile force in the tape (Fig. 9).

One of the primary design considerations assumes that the shovel might catch on a rock while digging a furrow on the lunar surface. To solve this problem, a recurrent set of expressions was written relating X and Y force components at the joint.

2. Mechanics

a. Antenna/Solar Panel Positioner (A/SPP). The A/SPP provides mounting, positioning, position indicators, and launch and landing locks for the planar array antenna and solar panel. The antenna support structure includes controllable axes for roll, elevation, and solar and polar rotations. The drive for each axis uses a solenoid stepper motor for power and a wire-wound potentiometer for position indication.

b. A/SPP 287580. Delivery of the spare solar axis drive has been delayed by the breakage of a connector pin during the low temperature test portion of flight acceptance testing. Pin breakage was the subject of an extensive investigation in October 1965. It was concluded then that the problem was caused by the thermal shock of a liquid nitrogen quench and the use of a mating connector of dissimilar materials which created excessive shearing forces on the pin. On the recent failure, preliminary investigation shows the following possible causes:

(1) The pin may have sheared by differential contraction from a thermal gradient between the connector halves caused by leakage of liquid nitrogen from the test fixture shroud directly onto the drive near the connector.

(2) A yellow color on 50% of the fracture surface indicates the possibility of gold plating entering a crack during manufacture. A metallurgical examination of the fracture will be made.

c. A/SPP 293275. The design verification test model (Fig. 10) has been assembled and is now undergoing tests. These functional tests will verify the adequacy of the design and establish limits of the A/SPP operational capabilities. The design verification testing consists of the following:

(1) Bearing friction torque tests in the unloaded condition.

(2) Bearing friction torque at ambient temperature with minimum and maximum bearing loads.

(3) Functional performance at maximum lunar loads.

(4) Backlash and spring rate at ambient temperatures.

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(5) Thermal vacuum endurance tests to demonstrate the capability to function for the required life in a simulated lunar environment.

(6) Post-thermal vacuum.

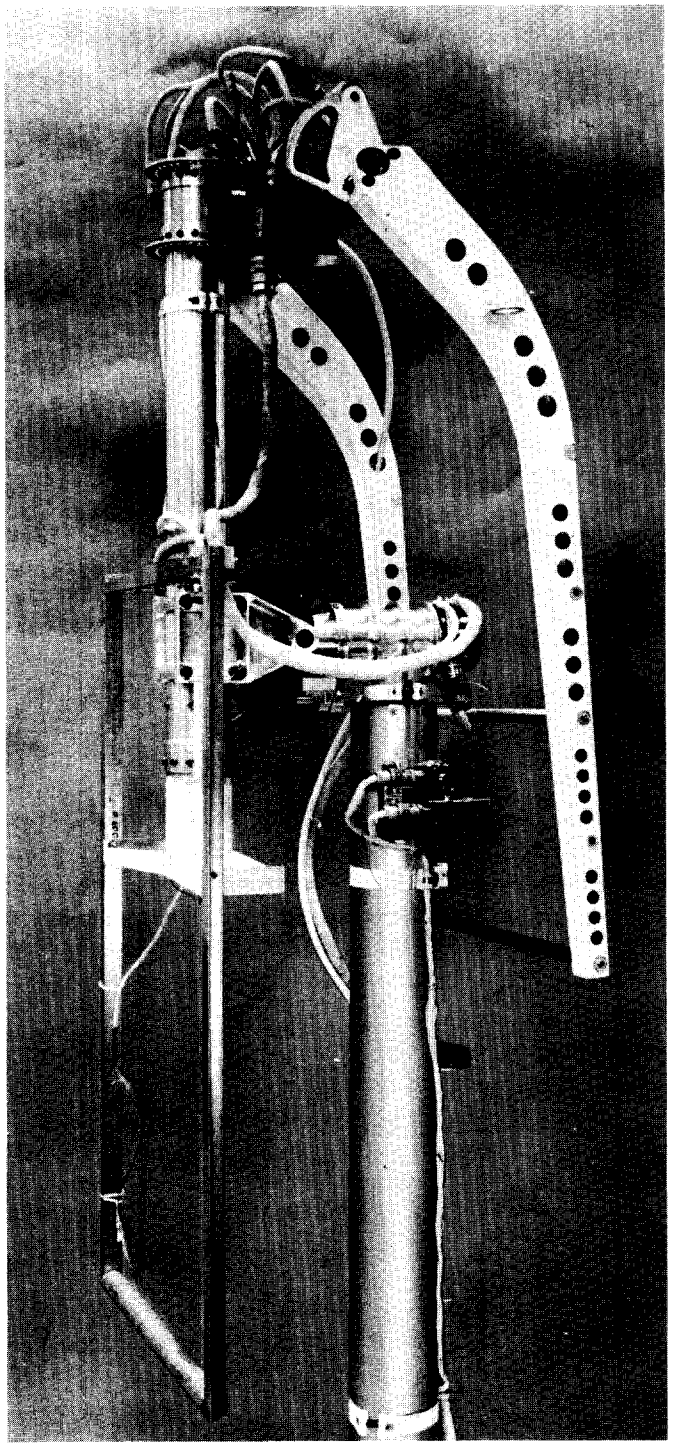

Fig. 10. A/SPP design verification test model

The design verification test model is now in post-thermal vacuum test. During this series of tests, connectors soldered to the roll and elevation drive housings were found to be loose. High temperature during the thermal vacuum test is believed to have softened the solder. This did not occur during high temperature qualification tests of the drives. The failure is under investigation.

No design verification test of the structure will be conducted since all structure was upgraded to show positive margins. Typical upgrading of the structure is revealed in Fig. 11, which is a closeup of the roll and elevation axes structure in the pin puller area. Note the high-strength steel adapter mounting bracket for the roll axis pin puller module. Fig. 11 also reveals upgrading of the elevation axis structure in the pin puller area.

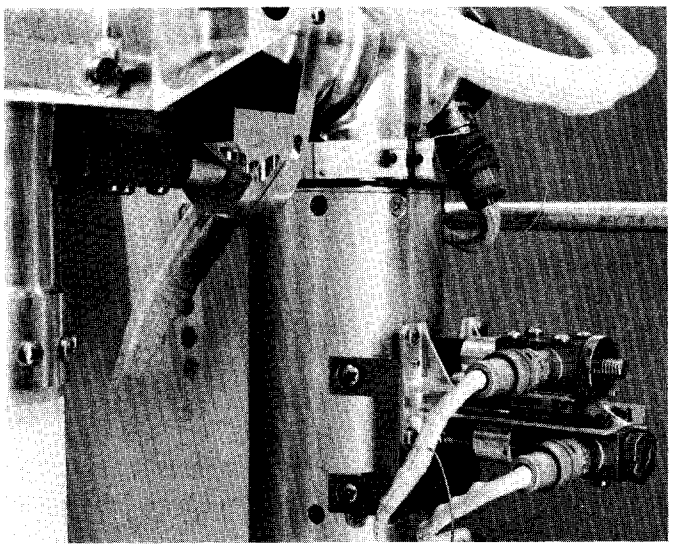

Fig. 11. A/SPP structurally improved pin puller tactics

Qualification of one of the four potentiometers undergoing tests at the components and materials Laboratory was completed in November. One unit failed under test because of excessive wiper wear. This failure is under investigation. The remaining potentiometers underwent the rotation in vacuum life test the latter part of November. All tests have been completed.

Studies were conducted to determine if it is possible to relock the roll, solar, and elevation axes (the polar axis is self-locking) after a lunar landing, so that a liftoff and translation experiment could be conducted. The relocking is essential because of the dynamic loads encountered on relanding.

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d. High-torque motor and transmission. During this reporting period, the qualification test, including the release of the qualification test report, was concluded. A summary of the tests and a comparison of required life and test life are included in this report.

The qualification test was performed on the polar drive having a 7.2:1 gear ratio, and the roll drive having a 289:1 gear ratio. Although the polar drive gear ratio was later increased to 14.2:1, late availability precluded its use in these tests.

The units were tested for electrical requirements and mechanical performance requirements before and after exposure to all significant pre-launch, launch, and lunar environmental conditions. These include shock, vibration, and thermal vacuum below 1×10^{-6} torr in the temperature range of -225 to 260°F. Both drives met the requirements throughout the life qualification test of 120,000 steps.

Extended testing to determine life beyond the minimum requirement was also performed. The roll axis drive motor failed after 14,900 steps beyond the minimum requirement of 120,000 steps. The failure was caused by a weld-joint break between the armature and one of the detent roller shafts. The armature and pin were sectioned, and microphotographs were taken. This revealed that the weld-joint penetration was not as deep as required. To give adequate strength to the assembly, a chamber was added to the armature, and a filler rod was used. This provides better control of depth of the weld.

The polar axis drive experienced a wearout failure after 80,000 steps beyond the minimum requirement of 120,000 steps. The output torque decreased to less than the lunar torque load of 5.0 in.-lb, when powered with the minimum power supply output at -225°F. After disassembly, parts inspection revealed that the dry film lubricant was completely worn off on the interfaces between the solenoid armatures and the sprocket-shaft, allowing metal-to-metal contact and subsequent wear of both the armatures and sprocket. The increased friction resulted in motor output torque decrease.

It has been observed on practically all motors tested, developmental as well as the two qualification units, that the dry film lubrication breakdown takes place rapidly after approximately 120,000 to 130,000 cycles. This loss of lubrication determines the life of the motor.

e. Thermal switches. Active elements of the *Surveyor* compartment thermal control are the nine thermal switches on compartment A and the six thermal switches on compartment B. These bimetallically-operated units either make or break a conductive path to fixed radiators as a function of compartment temperature to maintain temperature within a narrower band than the external temperature excursions.

During the report period, the type approval test review committee accepted type approval test results, including the supplemental aerodynamic heating test, and issued the concurring memorandum on unit flight qualification.

A revised vehicle installation procedure has been coordinated and implemented on SC-4. This revised procedure employs a mounting template for positioning switches relative to each other, thereby avoiding the time-consuming alignment problems on previous vehicles.

Requirements for vycor glass radiators for the alpha-scattering sensor head have been satisfied by reworking existing radiator assemblies used on the compartment A thermal switches. A request was received from JPL and LASR–University of Chicago, to study the possibility of mounting an aluminized vycor mirror on the top surface of the instrument. The large reduction in solar absorptance by use of the vycor mirror allows a much lower unit operating temperature permitting full lunar day operation, as well as increased unit reliability. The sensor head size is such that the compartment A thermal switch radiators were easily modified to fit the installation. The modification consisted of machining the radiator base to mate the sensor head top cover over a 6 in.^{2} area, plugging the existing radiator mounting holes, and providing new mounting holes. The entire task, from negotiation to final delivery (six articles required), was completed in the report period. The modified radiators are shown in Fig. 12.

f. Omnidirectional antenna mechanisms. Omnidirectional antenna mechanisms A and B use tapered fiberglass cloth tubes bonded to aluminum hinge bracket assemblies. They deploy and lock the antennae into operating position before the spacecraft separates from *Centaur*.

During the SC-3 spacecraft alignment, it was determined that the omnidirectional antenna A dipole alignment was incorrect, and the problem was traced to faulty manufacturing tooling. All previously delivered units are being reworked to correct the discrepancy. Reworked units for SC-3 and SC-4 have been redelivered to their respective spacecraft. The pedestal locking spike on the

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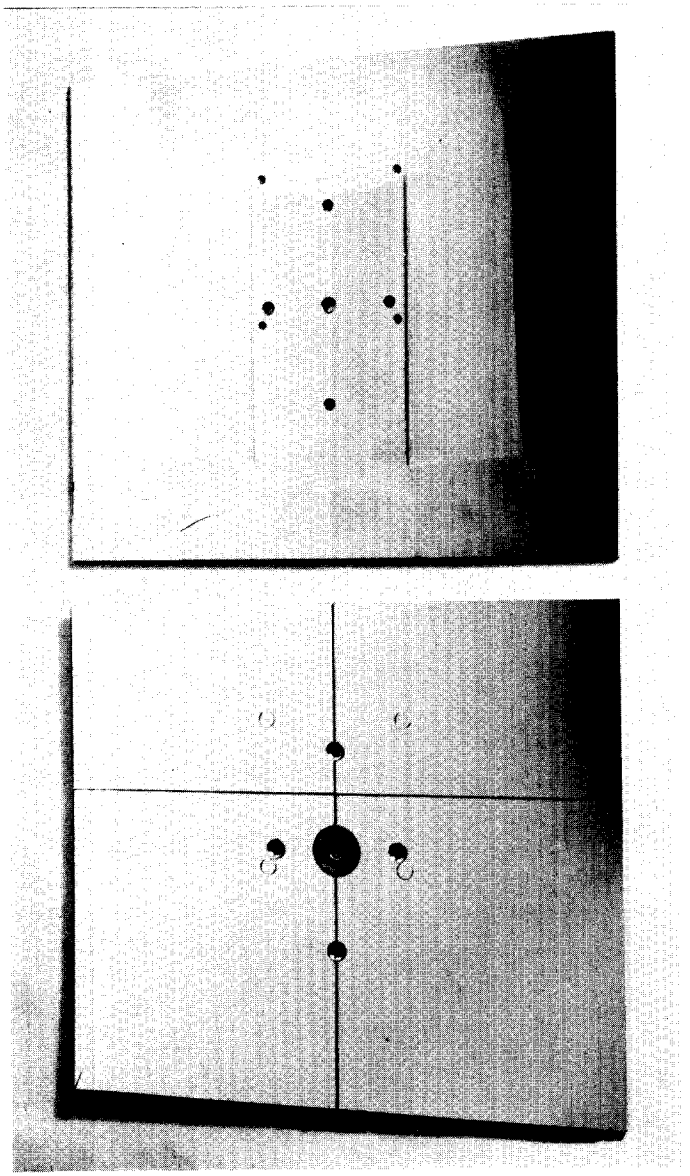

Fig. 12. Thermal switch radiators modified for use on alpha scattering sensor head

omniantenna A was also found to be inadequate to accomplish the desired alignment. The spike is now being redesigned and will be retrofitted on all the omniantennas starting with SC-3.

g. Separation sensing and arming device. Three separation sensing and arming devices for each spacecraft provide redundant actuation of the postseparation squib-firing circuits and vernier engine solenoid circuit after the spacecraft separates from *Centaur*. A set of units for SC-6 was delivered, and the SC-7 units are in flight acceptance test.

h. Pyrotechnic devices. A pyrotechnic review meeting for all SC-3 items, similar to a consent-to-ship meeting, was held in late November. The design fabrication, test, and quality status of all flight and spare pyrotechnics was reviewed and approved for SC-3.

Assistance was provided the launch operations department in updating report 227431, "A-21 Pyrotechnic Handling, Testing, and Installation Procedures," to reflect the latest configuration and techniques to be used in preparing SC-3 for launch.

Qualification and lot acceptance testing of squib 293162 was completed at the Hi Shear Corporation, and all squibs have been received at HAC.

i. Flight and test vehicle status.

SC-3 flight spacecraft. The spacecraft completed solar thermal vacuum and vibration testing, and, after a final weighing, balance and alignment, was shipped to AFETR. This was the first flight vehicle to receive both a previbration and postvibration alignment check. There was indication of slippage of several critical alignment conditions, but all except one were procedure problems. Initial installation of alignment target mirrors on the flight control sensor group (FCSG) was incorrect because of a grommet on the FCSG protective cover. This caused the FCSG to be mis-shimmed. With the cover removed and the alignment mirrors properly seated, the required shimming was identical to SC-1. In the future this cover will be removed prior to installation of optical mirrors to forestall future alignment difficulties.

Thrust chamber assembly 3 shifted approximately 5 min from the initially aligned position. This shift was probably caused by the fueling and defueling operation rather than from vibration forces. Movements of the same order have been experienced on SC-1 and SC-2 at AFETR after fuel loading.

An apparent shift of the retro-rocket thrust axis was detected, but further investigation revealed this shift to be the result of two improper installations of the alignment tool. Repeating the erroneous procedure gave the same position after vibration as before. Another problem was discovered in working with the dummy retro-rocket. The staking on the spherical bearings on the support struts have loosened to the extent that the alignment data are not repeatable. The retro-rocket has been returned to the supplier for rework.

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The only shift which was a result of a hardware problem was the leg 1 nitrogen jet. The bracket which holds the jet on the leg yielded, moving the jet out of tolerance. The bracket has been redesigned to correct this condition.

As a result of the problems on SC-3, the plan for SC-4 has been revised to include a complete pre- and post-vibration alignment to reestablish confidence in the procedures and the use of the tools.

Several ECAs were incorporated during a special upgrade to reflect the recommendations of the SC-2 failure review board. They included the following:

(1) ECA 113033 added a test access connector for verifying vernier engine solenoid connections.

(2) ECA 113038 inhibited gas jet operation by interlocking with the postseparation, sensing, and arming 22-v power.

(3) ECA 113039 optimized vernier propulsion system temperature data transmission by switching commutator modes.

(4) ECA 113041 added the EP-14 filter to reduce the noise on the regulated output current telemetry.

SC-4 flight spacecraft. The spacecraft was upgraded in preparation for solar thermal vacuum test. Included in the upgrade were several changes resulting from the SC-2 failure review board recommendations as discussed above for SC-3. Subsequent to the upgrade, the vehicle was returned to the environmental laboratory for solar thermal vacuum test.

SC-5 flight spacecraft. The spacecraft completed a limited ISCO and was returned to the assembly area for an upgrade period in which the harnesses were retro-fitted with the wiring changes introduced by the extended payload as well as ECA changes similar to those on SC-3 and SC-4. The structural upgrade to install the alpha scattering experiment and TV auxiliary viewing mirrors is not scheduled during this reporting period. The vehicle has been delivered back to systems test.

SC-6 flight spacecraft. This vehicle nears completion of final assembly and is scheduled for delivery to ISCO the fourth week in December (Fig. 13).

SC-7 flight spacecraft. This vehicle holds the lowest priority because of many mandatory changes on earlier vehicles. It is in the early phases of final assembly (Fig. 14).

G. Thermal Control

SC-3 solar thermal vacuum (STV) testing took place during October. Flat auxiliary TV viewing mirrors and a soil mechanics/surface sampler (SM/SS) mockup were installed prior to phase 1C testing. Subsequent data demonstrated that neither addition significantly affects spacecraft transit thermal control. Temperatures on the mirrors and the SM/SS mockup were as anticipated (see Table 2).

Table 2. Solar thermal vacuum test data for TV viewing mirrors and SM/SS

Item	Phase 1-C		Phase 2-C		Phase 3-C	
	Low intensity (87%), °F	High intensity (112%), °F	Low intensity (87%), °F		Nominal intensity (100%), °F	
TV viewing mirrors						
Large	10	40	14		25	
Small	24	56	24		42	
SM/SS	SM/SS did not operate in Phase 1-C		Heater off	Heater on	Heater off	Heater on
Auxiliary electronics radiation	−49	−20	−46	+17	−31	+23
Auxiliary electronics canister	−77	−55	−75	−70	−65	−60
Mechanism	−126	−109	−123		−117	

Vernier propulsion system oxidizer tank 1 cooled down at a greater than normal rate during phase 1C testing. The problem was traced to the superinsulation blanket, which was wrapped too tightly. After replacement, prior to phase 2C, the tank cooldown rate was normal.

Among several studies that were performed to support both the A-21 and A-21E spacecraft was a definition of the alpha-scattering instrument (ASI) sensor head thermal environment for transit and lunar surface operations. ASI compartment C transit and lunar operation constraints were developed, using a simplified compartment model, and a final analysis for the SC-3 and SC-4 flat auxiliary TV viewing mirrors indicated that mirror temperatures will be between −110 and 260°F for an entire mission (excluding a lunar night landing situation, after which the temperature would drop to −300°F). The JPL design for the SC-5, -6, and -7 solar panels was reviewed and compared with the SC-1, -2, -3, and -4 solar panel design. There were no noteworthy thermal differences from the solar panel

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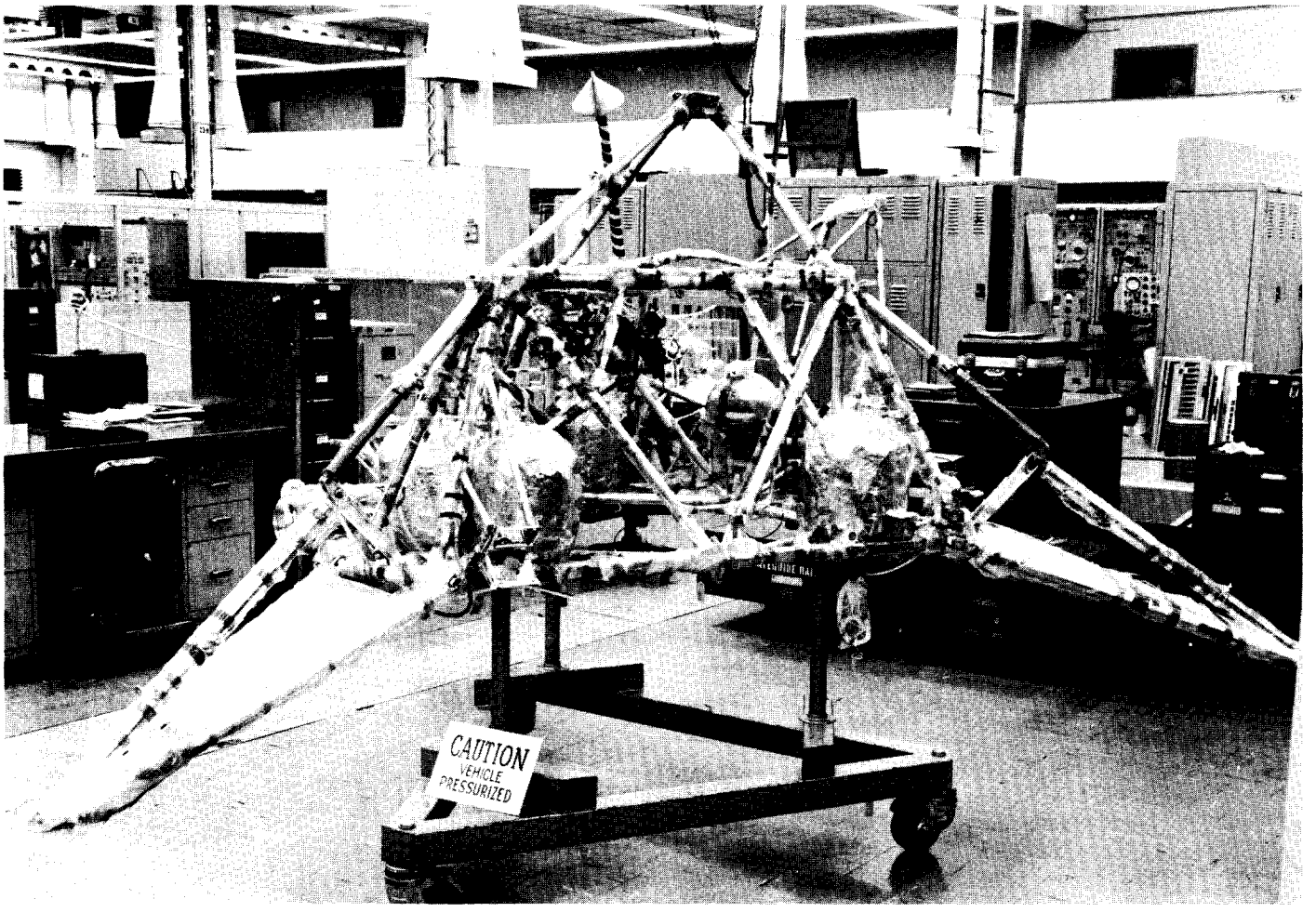

Fig. 13. SC-6 final assembly

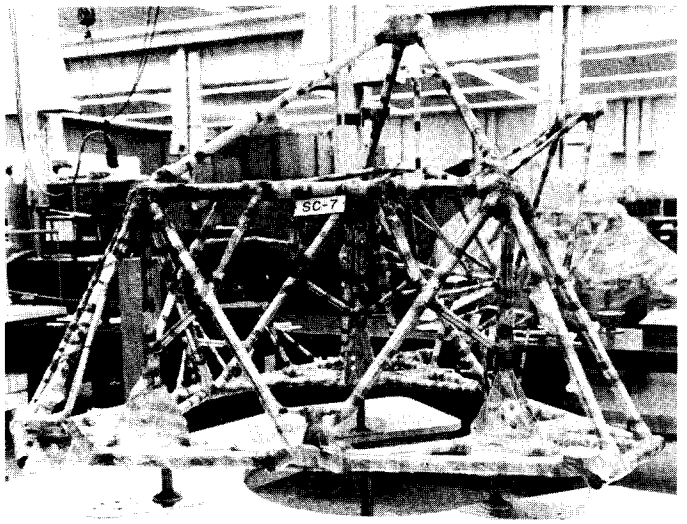

Fig. 14. SC-7 early assembly

point of view; however, the thermal finish for thrust chamber assembly 2 requires modification.

H. Spaceflight Operations

1. Fifth Lunar Day Operations

At 04:26:10 GMT (21:26:10 PDT, Friday, October 7), *Surveyor I* transmitted engineering data to DSS-51 (Johannesburg) on command. The objective in reactivating the SC-1 spacecraft was primarily to conduct approximately 12 experiments which were originally intended for SC-2. For the most part, these experiments were concerned with telecommunications and television. The first command sequence sent was to energize transmitter A, but there was no response. The next command sequence sent was to energize transmitter B. This sequence was successfully received by the spacecraft.

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The signal level at DSS-51 indicated approximately −142 dbm. An engineering commutator assessment was made and all subsystems looked promising. As a result of this assessment, it was decided to step the solar panel perpendicular to the sunline. When the command was sent to step the solar panel, the spacecraft signal was lost. It was recovered using nonstandard procedure SPAC 32 in the same manner that was used for the initial revival.

From 21:56 to 22:26 PDT, spacecraft data were intermittent. At 22:26 PDT, the spacecraft was acquired again, and a reassessment of the engineering commutators was made. Signal strength slowly degraded to −161 dbm. Shortly before 23:05 PDT, the bit rate was reduced from 137.5 to 17.5 bits/sec to minimize data dropouts. At 23:45 PDT, the command link was lost from DSS-51 until the end of the pass at 02:00 PDT (Saturday, October 8). There was no deep space network (DSN) coverage until 10:00 PDT when DSS-42 (Canberra) visibility began.

The spacecraft was then acquired with a signal strength of −158 dbm but data lockup and a command link could not be achieved until 14:03 PDT. At this time, the spacecraft was reassessed, and the bit rate was increased from 17.5 to 137.5 bits/sec. The signal strength had increased to −148 dbm. At the time of acquisition, it was noticed that the auxiliary battery mode was on; main battery voltage was 9.5 v; and EP-2 (unregulated bus voltage) was 26.5 v. The main battery mode was commanded on, and the spacecraft signal was lost. The spacecraft was recovered shortly thereafter and indicated that it was in the main battery mode, high current off, with both batteries in poor condition. Terminal voltage of the auxiliary battery was reading 0 v with a main battery pressure of 0 psia.

At 20:55 PDT Saturday, command 0307 (optimum charge regulator bypass) was sent from DSS-51 to improve the battery situation. The spacecraft signal strength dropped from −148 to −153 dbm, and spacecraft data lockup was not attainable. Additionally, the command link was lost for the remainder of the DSS-51 pass, which lasted until 03:20 PDT (Sunday, October 9). There was no DSN coverage from 03:20 until 11:00 PDT.

At 13:25 PDT, DSS-42 obtained an uplink with the spacecraft, which provided good data with a signal strength of −147 dbm. An evaluation of the engineering commutator and the power subsystems was made. This interrogation indicated that the spacecraft was good or better than it was on Saturday (October 8), and the solar panel was stepped in the plus direction. Data lock was lost for a few minutes and then reestablished.

At about 16:21 PDT, an attempt was made to change to mode 7. When the camera power-on command was transmitted, the spacecraft signal was lost. The spacecraft was brought back up again within a few minutes, and the survey camera heaters were turned off. Again the command camera power-on signal command was given and again the spacecraft signal was lost. The spacecraft was promptly reacquired. It was then decided to turn off all power loads including the spacecraft carrier. The command camera power-on command was given and accepted by the spacecraft. The processing system was then brought up slowly, and mode 7 data were successfully obtained, although with intermittent sync and data dropouts.

An engineering commutator interrogation was made, and the traveling-wave tube (TWT) filaments were energized for 5 min. A test of the transfer switch capability was accomplished. The antenna/solar panel positioner polar axis was stepped, to no avail. It was then decided to take TV pictures. This sequence for TV pictures consisted of the following:

(1) Turn off signal processing and transmitter.

(2) Turn off nonessential loads.

(3) Enable nonessential regulator.

(4) Transfer switch B to higher power.

(5) *Surveyor* camera power on.

(6) Transmitter B filament power on.

(7) Set lens to 100 mm focal length.

(8) Transmitter B low power on.

(9) Narrow-band voltage-controlled crystal oscillator on.

NOTE: A 10-sec time interval was allowed between commands, up to the TV start frame command, to allow the battery to recover from transients.

At 01:30 PDT (Monday, October 10), the TV sequence was initiated for the first time. The above TV sequence was initiated again at 03:20 and 04:11 PDT. All attempts failed. DSS-51 lost contact while attempting to take TV pictures during the setting of the Sun on the Moon. DSS-61 (Madrid) could not acquire the spacecraft during its visibility period. There were no further efforts to work the spacecraft.

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The original magnetic tape was received and dubbed on October 18. A strong 70-Hz carrier was observed both during the dubbing and subsequent playback of the dubbed tape. The demodulated output showed no 200-line mode video present. There was, however, a 40-sec period starting at about day 283, 09:10:15 GMT, during which noise was seen at the demodulator output.

On October 19, a more detailed analysis of the video was made using a spectrum analyzer to observe both the 70-kHz predetection signal and the demodulated output. This check was made to see if any periodic components indicating sync pulses were present. None was found in the two areas specifically checked; area 1, 283:09:10:17 GMT, for approximately 30 sec and area 2, 283:08:35:00 GMT, for approximately 80 sec.

A 70-kc carrier was inserted by the command and data-handling console (CDC) equipment until the 70-kHz carrier from the spacecraft was above a certain amplitude threshold. The spacecraft 70-kHz carrier was recorded at a level half that of the CDC-inserted 70-kHz carrier. The level and frequency of the predetection 70-kHz carrier were monitored. At all times except those shown below, the CDC-inserted carrier was observed. Its frequency was 70.175 kHz ±20 Hz. At all times listed below, the 70-kHz carrier dropped to about one-half amplitude:

Time (GMT)	Duration (sec)
08:30:32	Less than 1
08:30:46	4
09:03:22	Less than 1
09:10:15	40
09:31:56	21
09:53:42	25

The total time of observation was from 08:30:00 to 10:00:00 GMT. This time covers the TV sequences as indicated in the DSS-51 log sheet. The demodulated video output at the above-listed times showed no evidence of any 200-line mode video.

2. Sixth Lunar Day Operations

On 02:33 PST on Tuesday, November 8, one-way lock was obtained with *Surveyor I*. Nonstandard procedure SPAC 32 was again employed to revive the spacecraft, which occurred immediately upon ground command. During the *Surveyor I* sixth lunar day operations, the Mars station 210-ft dish antenna was used for commanding the spacecraft via DSS-11 (Goldstone). At 03:00 PST, two-way lock was obtained with a signal strength of −131 dbm. An engineering interrogation was made on modes 1, 2, 3, 4, 5, and 6, and all subsystems look promising.

As a result of this assessment, it was decided to go to mode 7 to get TV ID. When the transponder power-off command was given, the spacecraft signal was lost but was recovered using nonstandard procedure SPAC 32. At 03:53 PST, TV ID was obtained with a returned signal strength of −138 dbm. Based on the fifth lunar day performance, all excessive loads were turned off before stepping the camera. At 04:12:40 PST, transmitter B was brought up in low power with a bit rate of 137.5. Spacecraft mode 7 telemetry was received at a signal level of −132 dbm. At 04:29:23 PST, the receiver was continuously in and out of lock, thereby forcing the spacecraft to be shut down. Transmitter B was turned back on at 04:30:32 PST but had to be shutdown at 04:34 PST due to the requirements of the *Lunar Orbiter* mission. It was not until 04:57:44 PST that transmitter B was turned back on, and one-way lock was achieved with a signal strength of −137.7 dbm.

At 05:26 PST, the first TV start frame was commanded, with no response, and again at 05:42 PST TV start frames were sent with no response. At 06:30 PST, transmitter A was turned on, and the spacecraft was then acquired with a signal strength of −153 dbm, which slowly degraded to −160 dbm. It was then decided to switch to transmitter B. A signal level of −147 dbm was received, which, when the signal processing was turned on, decreased to −156.5 dbm. The spacecraft was turned off for 1½ hr to allow the battery to recharge. At 08:31:20 PST, transmitter B was turned on, and an engineering assessment was made in mode 5 at a bit rate of 137.5 bits/sec. The received signal level was −131 dbm.

At 09:00 PST, transmitter A was turned on. The receiver was in-lock with a signal level of −149 dbm. At this time, all nonessential loads were turned off to allow the battery to charge in preparation for TV sequences. At 09:56 PST, transmitter A was commanded to high power with a total of six start frames; no indication of video was received. It was decided to go to low power with transmitter A; there was still no response. At 10:32:39 PST, transmitter B was commanded on in low power. The receiver was in-lock with a signal strength of −149.5 dbm. At 10:49:53 PST, the spacecraft was again turned off due to a *Lunar Orbiter* mission.

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At 12:12 PST, transmitter B was turned on in preparation for high-power TV picture taking. The 560 Hz SCO was commanded on at 17.2 bit/sec with no response. The 960 Hz SCO was commanded on and the signal strength decreased from −148.5 to −155 dbm with no bit stream obtainable. At 12:42:42 PST, the spacecraft was commanded off to conserve power, and at 13:17:35 PST transmitter B was turned back on. The receiver was locked up with a signal strength of −149 dbm which slowly degraded to −158 dbm. At 13:27:05 PST (November 8), the spacecraft was turned off, ending the sixth lunar day revival attempt.

Generally, the *Surveyor I* performance was comparable to that of the fifth lunar day operations. Transmitter A was exercised for the first time since the second lunar day. Transmitter B continued to evidence the low signal level failure mode noted in transmitter A during the second lunar day; that is, an unpredictable and occasional drop of about 17 db in signal level. It was assessed that the video channel feeding transmitter B was inoperative, and that the start frame command could not initiate transmitter A high-power operations. Consequently, TV pictures could not be obtained. Two-way doppler tracking was completed, and DSS-14 was checked and verified operational.

Preliminary analysis of the sixth lunar day signal processing of all six commutators showed that many of the commutator 2 readings were substantially different (sometimes zero or full scale) than other commutator readings of the same signal. Reduced usage of commutator 2 will therefore be scheduled for the SC-1 seventh lunar day revival.

I. Deep Space Network Test, Operations, and Training

1. Modification Kit Installations

Extensive command and data-handling console (CDC) upgrade was accomplished at all DSN stations. The most extensive modification, ECA 112021-3 (designed to facilitate immediate switching between 200- and 600-line TV modes without recalibration), was initially installed, proofed, and tested at Goldstone prior to installation at the overseas stations. Corrected engineering and revised procedures were then provided to facilitate the modification installation at the overseas stations. Table 2 is a tabulation of the modification kits installed at DSS-11, -42, and -51 between missions B and C through November 28. Modifications were also performed at DSS-61, -71, and -72 during this time.

2. CDC Patchboard/Overlay Control

Considerable effort was expended to improve and streamline the control cycle for updating the CDC mission patchboards and telemetry overlays. Primary concerns were:

(1) Reducing cost.

(2) Optimizing internal procedures to minimize change cycle time.

(3) Facilitating the information flow and updating problems on-site.

Specific steps included the following:

(1) Changing format and reducing size of the patchboard/overlay control specification.

(2) Eliminating redundancies and conflicts between the specification and the DSIF detailed operating procedures (DOPs).

(3) Revising the format and technique of overlay chip manufacture to reduce significantly the amount of aluminum material involved.

(4) Precutting the chips before transmittal to the stations.

These steps, coupled with the previously instituted technique of automated scale manufacture (computer-driven plotter), are expected to provide flexibility to cope with extensive last-minute changes.

3. Special DSIF Station Test, Operations, and Training

Command data handling console (CDC) upgrade was accomplished at Madrid (DSS-61), Ascension (DSS-72), and AFETR (DSS-71) during this period. Modifications included some that were not specifically required because of the reduction in equipment at Cape Kennedy and Ascension, but were installed in keeping with the philosophy of maintaining all DSIF CDC units interchangeable to the maximum practical extent. Table 3 is a tabulation of the modification kits installed at DSS-61, -71, and -72 between mission B and C, through November 28.

Training of DSN personnel at DSS-71 commenced during this report period and continued at DSS-61.

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Table 3. Modification kits installed at DSS-61, -71, and -72

ECA	Subject	DSS-61	DSS-71	DSS-72
111991-1	Video logic and gating unit modification	X	X	X
112000-1	Removal of error detection lamp	X	X	X
112134-3	Corrections to ECA 112134-1	X	Does not apply	Does not apply
112134-4	Corrections to ECA 112134-2	X	X	X
112169-3	Punched tape reader restrictor addition	X	X	Does not apply
112681-3	TV vertical sweep modification	X	Does not apply	Does not apply
133204-1, 2,6,7,9,10	Addition of displays to SOC console	Does not apply	X	X
133204-8	SOC console cable correction	X	X	X
133204-9	Corrections to ECA 133204	X	Does not apply	Does not apply
133204-10	Corrections to ECA 133204	X	Does not apply	Does not apply
133269-3	TV dc level shift modification	X	Does not apply	Does not apply
133268B	Command generator power supply modification	X	X	X
133281	Patchboard verification tester modification	X	Does not apply	Does not apply
133298	Wire deletion (telemetry converter)	X	X	Does not apply
133295	Test point addition (demodulator)	X	X	Does not apply
133306	Fuse change (time and data buffer)	X	X	X
133307	Command generator modification	X	X	X
133313	Add resistor in telemetry patch panel	X	X	X
133320-1	TV generator and video logic gating unit	X	Does not apply	Does not apply

J. Operational Support Equipment

1. Command and Data-Handling Console CDC-9

CDC-9, the last of the presently scheduled operational support equipment, was shipped from HAC on November 23, 1966, arriving at AFETR on November 26. After final checkout, the equipment will be operational early in December 1966. The following paragraphs describe significant open items remaining.

2. CDC-9 Video Processor

This unit was held to verify the released drawings and as a model in the preparation of engineering change analysis (ECA) 112020-3 modification kits. Following a successful field test of the video processor at DSS-11 (Goldstone) during the week of December 2, it is expected that all DSIF CDCs will have the new design video processor installed in time to support SC-3.

3. ECA 133308

ECA 133308, "Elimination of Ground Loop to Inhibit Noise Pickup in the TV Subsystem" was shipped as a modification kit to be incorporated on-site. Its purpose is to remove a ground loop condition that has been causing a 60-Hz ripple voltage on the CDC TV displays.

4. Channel Selectors

CDC-9 is short some channel selectors needed to support SC-3 tests at AFETR. The latest promised delivery date of January 1967 by the vendor (Electro-Mechanical Research Company) makes the test support problematical, and a major attempt is being made to shorten this delivery time.

5. ECA 133326

This modification kit for the buffer addition to the programmed automatic telemetry evaluation PATE distribution unit was expedited to meet a December 27, 1966 installation schedule date at AFETR. The purpose of this modification is to add line driver-amplifiers so that the CDC decommutator memory output is sufficient to operate the PATE distribution unit under all loading conditions now experienced. This modification will also be made to all STEA CDCs.

6. New Tape Readers/Spoolers

Change order 106 to JPL contract 950056 was negotiated and full authority to proceed was given. Vendor fabrication of new tape readers and spoolers of an advanced model proceeded on schedule. Tests were completed at DSS-11 (Goldstone) on the test model reader, and a test report was submitted to JPL ("Tape Reader Command and OSDP Bit Error Rate Test," SSD 64257R, dated December 15, 1966). Field checkout of the first production article for specification compliance is also being conducted at DSS-11. Successful completion of this testing (approximately December 15) will permit release of the full production order and delivery of the reader/spoolers.

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7. Modification Kit/ECA Status

Engineering change accomplishment status records were completed and published on November 26, 1966 in the CDC management plan, document 3023333. These records allow for accurate evaluation of configuration baselines for all CDC equipment (both DSIF and STEA). The most significant changes for this mission affected the TV subsystem.

K. Launch Vehicle Integration

1. Surveyor Mission B

Atlas/Centaur AC-7, with its *Surveyor II* payload, was launched from ETR complex 36A on September 20, 1966, after prelaunch operations which were essentially in accord with planning data in SPS 37-40, Vol. I.

The spacecraft was lofted into a lunar intercept trajectory in a single-burn, direct-ascent mission. Through the *Centaur* retromaneuver, the flight was normal, as indicated by Table 4.

Table 4. Comparison of AC-7 Preflight and Actual Events Time

Event	Preflight time (sec)	Actual time (sec)
Programmer start; 2-inch rise (Liftoff)	T + 0	T + 0
Booster engine cutoff (BECO)	T + 142.72	T + 142.2
Jettison booster package	T + 145.8	T + 145.3
Jettison insulation panels	T + 176.7	T + 176.0
Jettison nose fairing	T + 203.7	T + 202.9
Sustainer engine cutoff; vernier engine cutoff; start *Centaur* programmer (SECO)	T + 235.35	T + 235.1
Separate *Atlas–Centaur*	T + 237.2	T + 237.0
Centaur main engine ignition	T + 246.8	T + 246.6
Centaur main engine cutoff (MECO)	T + 683.4	T + 686.3
Preseparation arming signal; extend landing gear signal	T + 710.8	T + 710.6
Unlock omniantennas signal	T + 721.4	T + 721.1
High-power transmitter-on signal	T + 741.8	T + 741.6
Separate spacecraft	T + 752.8	T + 752.6
Energize power changeover (end of LV mission)	T + 1242.9	T + 1242.9

a. Launch countdown. The launch countdown on September 20, 1966, proceeded routinely until activation of the *Atlas* telemetry battery at T-260 minutes resulted in a low open-circuit voltage, which reflected a decaying trend. The battery was replaced with no impact on the count which proceeded normally until a scheduled 15-minute hold at T-5 minutes. The hold was extended 7 minutes to investigate an apparent low-temperature indication within the *Centaur* H_2O_2 system. The temperature was found correct, and the count was resumed. At T-115 seconds, it was necessary to hold and recycle to T-5 minutes because the *Atlas* LOX boil-off valve did not close at the start of the flight pressurization sequence. After the valve was exercised several times, and proper closure was obtained, it was decided to resume the count as soon as the *Atlas* LOX supply, which had fallen below the acceptable level, could be replenished. At this time, it was discovered that the automatic LOX topping system was malfunctioning, and manual operation was employed. After a hold of 26 minutes, the count was picked up at 07:27 EST, even though the *Atlas LOX* was still below the proper level, because the launch window for the day closed at 07:32. The flight level was reached at approximately T-3 minutes, and the count continued down to liftoff, which occurred at 07:31:59.8 EST. Atmospheric conditions on launch day were favorable. Surface winds were 6 knots from 190 deg with unrestricted visibility of 10 miles. Surface temperature was 79°F with relative humidity of 93% and a dew point of 77°F.

b. Launch vehicle performance. Launch vehicle system performance was generally nominal throughout its portion of the flight. Significant deviations were:

(1) Following *Atlas* booster engine cutoff (BECO +8.4 seconds), a high-roll oscillation of 4.3 deg/sec amplitude (peak-to-peak) was observed. No explanation for this anomaly has been found, but it is currently being investigated. At this time, the pitch and yaw rate amplitudes were 1.8 deg/sec and 1.9 deg/sec peak-to-peak, respectively, all three experiencing an approximate frequency of 0.77 Hz and lasting for not more than 4.5 sec. The high roll acceleration at this time was 13.9 deg/sec/sec and was not seen in previous flights.

(2) Significant variation between expected and recorded data on the *Atlas* booster LOX regulator reference pressure and sustainer gas generator discharge pressure were noted and are being investigated.

(3) After *Centaur* main engine start, system pressures stepped up to flight level within two seconds. C-1 system pressure was steady at 1072 psia through the engine firing. C-2 system pressure, however, underwent a gradual decay from 1080 psia at main engine start to 1030 psia midway through the engine firing and then gradually recovered to 1080 psia prior to main engine cutoff. The reason for this is not yet known.

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(4) After powered flight, a higher-than-normal *Centaur* current was experienced during and after squib firing for spacecraft separation. This anomaly which occurred after *Surveyor* electrical disconnect is being carefully analyzed.

(5) Telemetry tape received at JPL shows no data from three of the five accelerometers which measure spacecraft vibration during the boost phase.

c. Guidance accuracy. The launch vehicle accurately injected the spacecraft into the prescribed lunar intercept trajectory. Calculations show that the spacecraft would have missed the original targeted point by only 142 kilometers without a midcourse correction. The "miss only" injection error was 1.015 meters/sec and the "miss" plus "time-of-flight" error was 4.44 meters/sec. (both based on a scheduled midcourse maneuver 20 hr after injection). A total midcourse correction of 9.587 meters/sec was selected to accommodate a change in the targeted lunar landing site and optimize such terminal parameters as arrival time, main retro-rocket burn-out velocity, and vernier propellant margin.

d. Spacecraft/launch vehicle separation. A satisfactory separation was attained as evidenced by telemetry data which showed first motion of all three separation springs occurred simultaneously followed by nearly-identical spring extension slopes. Spring-induced tip-off rates were negligible. *Centaur* vehicle rates, immediately prior to separating the spacecraft were +0.12, +0.15, and +0.03 deg/sec in pitch, yaw, and roll respectively, or a total vector sum of 0.19 deg/sec. *Centaur* rates after spacecraft separation were +0.13, +0.14, and +0.04 deg/sec in pitch, yaw, and roll.

e. Centaur retromaneuver. The mean *Centaur–Surveyor* separation distance, 5 hr after injection, has been computed to be 730 km, compared to a minimum required distance of 336 km. The calculated nearest approach to the moon by the *Centaur* tank was 5675 km.

On December 3, 1966, a portion of the AC-7/SC-2 nose fairing was recovered on a Florida beach. Examination disclosed a hole through the fairing in a location which would have been adjacent to spacecraft vernier engine No. 3. An analysis of the recovered section is being performed by General Dynamics/Convair. At this time, there is no evidence that the damage has any connection with the failure of the SC-2 vernier engine system.

2. AC-9 Flight

The *Atlas/Centaur* vehicle AC-9 was successfully launched on a two-burn mission from ETR complex 36B on October 26, 1966. The *Centaur* with a 1750-pound mass model of *Surveyor* was injected into a nominal 90-nautical mile parking orbit. After about a 24-minute orbital coast, the *Centaur* engines were restarted, and the mass model was placed in a simulated lunar transfer trajectory.

The AC-9 flight was the final R & D flight in the *Centaur* program and the second of two vehicles designed to demonstrate an engine restart capability in space. This was the second launch attempt for this mission. The first attempt on October 25 was scrubbed because of a malfunction in both *Centaur* engine LOX pump inlet temperature-measuring circuits. The problem was resolved and the test rescheduled for a T-0 of 0600 EST on October 26. Liftoff on October 26 was 12 minutes later than planned because of a problem in destructor arm circuitry at T-2 minutes in the count. The count, which proceeded to T-90 seconds before a hold was called, was recycled to T-5 minutes to incorporate the required fix. Since the spacecraft did not contain propellants, the launch ladder circuit was modified to permit launch with a *Surveyor* safe indication. The count was resumed, and no further difficulty was encountered through liftoff.

a. Launch vehicle performance. The inflight performance of the *Atlas* booster was less than nominal because of a pneumatics malfunction which resulted in early sustainer engine cutoff with approximately 3 sec of burnable propellant remaining. The *Centaur* first burn and coast phases were normal and acceptable in all respects, including a first burn 10.3 seconds longer than predicted to compensate mainly for the short *Atlas* burn. Second ignition occurred as planned after a 24-minute coast phase and a totally-acceptable *Centaur* second burn was obtained. All mark events occurred as programmed, and the spacecraft was injected into a simulated lunar transfer trajectory, and the spent *Centaur* stage was satisfactorily retromaneuvered out of the spacecraft orbit. There was a close correlation between preflight estimates and actual measured times of occurrence of major flight events as evidenced in Table 5. Significant achievements of the flight were the successful coast phase propellant management of a cryogenic system and the subsequent restart of the *Centaur* engines.

b. Spacecraft/launch vehicle separation. Data from extensiometers on the separation spring assemblies indicate all three separation latches actuated within one msec of each other. The three separation springs operated nor-

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Table 5. Sequence of events

Event	Preflight estimated time (sec)	Actual time (sec)
Two-inch motion (Liftoff)	T + 0	T + 0
Booster engine cutoff (BECO)	T + 142.8	T + 143.1
Jettison booster package	T + 145.9	T + 146.6
Jettison insulation panels	T + 176.8	T + 177.3
Jettison nose fairing	T + 203.8	T + 203.8
Sustainer engine cutoff (SECO) by propellant depletion	T + 235.6	T + 229.1
Atlas/Centaur separation	T + 237.6	T + 231.3
First *Centaur* main engine start	T + 247.1	T + 240.5
First *Centaur* main engine cutoff	T + 573.5	T + 577.7
Second *Centaur* main engine start (MES 2)	T + 2034.8	T + 2034.7
Second *Centaur* main engine cutoff (MECO 2)	T + 2143.4	T + 2140.9
Separate spacecraft	T + 2210.8	T + 2210.9
Start 180-deg turnaround	T + 2215.8	T + 2216.0
Start retrothrust	T + 2450.8	T + 2450.9
Stop retrothrust	T + 2700.8	T + 2701.4
Energize power changeover switch	T + 2800.8	T + 2800.8

mally and yielded almost identical stroke versus time data. The DSN reported a spacecraft tumble rate of 0.5 deg/sec, well below the allowed 3 deg/sec, shortly after separation.

c. Centaur retromaneuver. The separation distance between *Centaur* and the *Surveyor* mass model five hours after separation was computed to be approximately 1194 km. The *Centaur* stage continued in a highly-elliptical orbit with an apogee of approximately 189,393 nautical mi, a perigee of approximately 92 nautical mi, and a period of 8.9 days; and it is reported to have re-entered Earth's atmosphere on November 14, 1966.

d. Guidance accuracy. The launch vehicle injected the spacecraft into a simulated lunar transfer trajectory which would have resulted in lunar impact approximately 630 nautical mi from the prelaunch aiming point. Approximate injection errors were 6.4 m/sec for "miss only" and 8.6 m/sec for "miss" plus "time-of-flight" (both based upon a spacecraft midcourse maneuver being performed 20 hr after injection). These corrections were both well within the three-sigma specification of 50 m/sec for "miss" plus "time-of-flight."

3. *Surveyor* Mission C

The launch vehicle (AC-12) for *Surveyor* mission C has been erected on ETR complex 36-B. The *Atlas* arrived on November 17, 1966 after a 9-day overland journey from San Diego and was erected on November 22, 1966. The *Centaur* arrived by air on November 15, 1966 and was erected on November 23, 1966. Preparations for a planned first mate with spacecraft SC-3 on January 6, 1967 proceeded without significant problems.

Available time did not permit cycling the spacecraft, SC-3, through combined systems test at San Diego, and it was airlifted directly to ETR on December 11, 1966. Following its arrival, and during the J-FACT preparation period, problems resulted in a decision to return the spacecraft to El Segundo for rework and further testing. Joint spacecraft/launch vehicle planning data for *Surveyor* mission C and following missions have not yet been re-established.

4. Interface Requirements

Negotiations on several proposed changes in the spacecraft/launch vehicle interface requirements have delayed the publication of Revision 3 to *Surveyor* Project Document No. 1. Important changes will cover requirements for:

(a) Incorporation of additional toxic gas detection equipment for vernier engine propellant leak monitoring.

(b) New ascent profile heating for parking orbit missions.

(c) Instrumentation redefinition to provide for additional measurements of the dynamic environment.

(d) Implementation of formal GSE safety checkout procedures.

(e) Addition of inspection ports in the spacecraft/launch vehicle forward adapter support ring and diaphragm for verifying cleanliness of the upper surface of the sterilization bulkhead after encapsulation.

5. Interface Control Drawing

JPL drawing 142800 is now the official *Surveyor/Centaur* interface control drawing replacing General Dynamics/Convair drawing 55-00050. Interface details shown by the JPL drawing are described in SPS 37-40, Vol. I, pp. 45 and 46.

6. Launch Vehicle Payload Capability

During this reporting period, the prime ascent mode was changed for *Surveyor* mission D to a parking-orbit mode and for mission E to a direct-ascent mode. This necessitated switching *Centaur* stages for these two missions.

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A summary of current launch vehicle payload capability for future *Surveyor* missions, as of approximately December 1, 1966, is presented in Table 6.

The current status shown in Table 5 reflects the switching of *Centaur* stages and ascent mode between missions D and E.

Table 6. Summary of current launch vehicle performance[a]

Mission	Flight	Payload	Ascent mode	Minimum burn-off wt, lb[b]	*Centaur* jettison wt, lb	Payload capability, lb	S/C wt, lb	Margin, lb
C	AC-12	SC-3	Parking orbit	6494	4088	2406	2292	+114
D	AC-11	SC-4	Parking orbit	6453	4112	2341	2282	+59
E	AC-13	SC-5	Direct	6539	3975	2564	2210	+354
F	AC-14	SC-6	Parking orbit	6732	4107	2625	2210	+415
G	AC-15	SC-7	Parking orbit	6747	4107	2640	2205	+435

[a]As reported in GDC 63-0495-43, "*Centaur* Monthly Configuration, Performance and Weight Status Report," December 21, 1966, unclassified.

[b]"Minimum BOW" equals "Nominal BOW" less the "Performance Reserve" of 175 lb for direct ascent missions and 195 lb for parking orbit missions.

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